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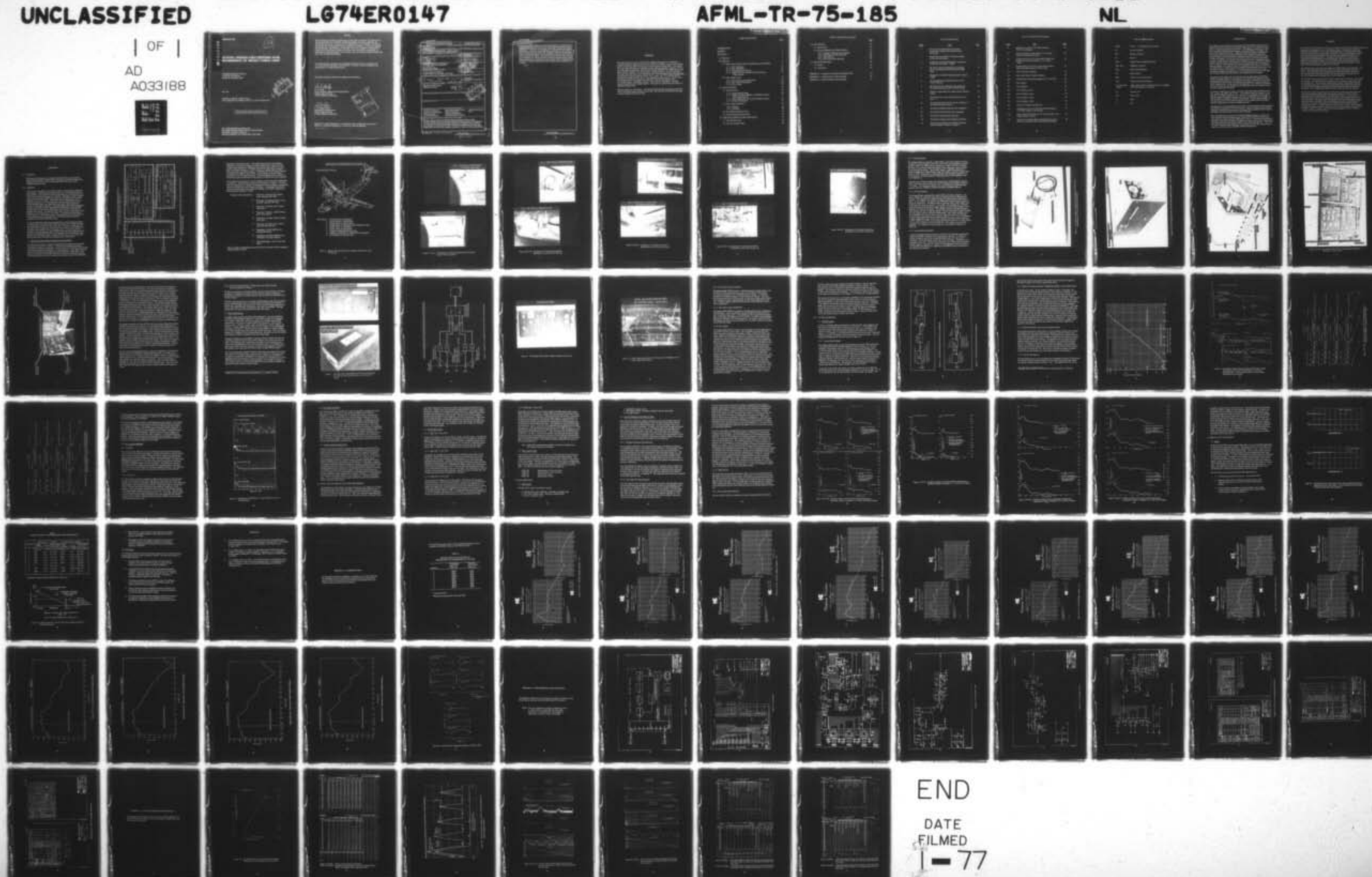

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ACOUSTIC EMISSION STRUCTURE-BORNE NOISE MEASUREMENTS ON AIRCRAFT DURING FLIGHT

LOCKHEED-GEORGIA COMPANY
86 SOUTH COBB DRIVE
MARIETTA, GEORGIA 30063

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TECHNICAL REPORT AFML-TR-75-185
FINAL REPORT FOR PERIOD 20 MAY 1974 - 29 NOVEMBER 1974

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AIR FORCE MATERIALS LABORATORY
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G. L. Hardy
Project Engineer, Aeronautical Systems Branch
Systems Support Division
Air Force Materials Laboratory

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AIRPLANE NOISE	AIRCRAFT INSPECTION	
NOISE MEASUREMENT	NONDESTRUCTIVE TESTING	

20. ABSTRACT (Continue on reverse side if necessary and identify by block number)

This joint program between the Lockheed-Georgia Company and the Air Force Materials Laboratory was concerned with measuring the structure-borne noise background in a large aircraft during four test flights to determine the feasibility from a signal-to-noise standpoint of using acoustic emission techniques to monitor structure during flight. Measurements were made in the frequency range from 0.1 MHz to 2.0 MHz using off-the-

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20. ABSTRACT (Concluded)

shelf acoustic emission transducers and 40-db preamplifiers in conjunction with a Lockheed-designed spectrum analyzer and a pulse code modulation data system. Calibration and flight data were recorded on the flight instrumentation magnetic tape recorder. Transducers and preamplifiers were installed at nine locations on the aircraft including the wing, pylon, main landing gear well and the empennage. A commercial flaw locator system was also installed to monitor the center wing lower surface during the four flights. The results showed that structure-borne noise can vary considerably over the aircraft structure and that a suitable acoustic emission signal-to-noise ratio can be achieved at most locations within the frequency range 0.5 MHz to 0.75 MHz, although the noisiest areas may require operation up to 1.0 MHz.

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FOREWORD

This report describes the tests and results of a Program conducted at the Lockheed-Georgia Company to determine the structure-borne noise endemic to an acoustic emission system for detecting stable crack growth in aircraft structure during flight. The program was a joint effort between the Lockheed-Georgia Company, Marietta, Georgia, and the Air Force Materials Laboratory under Contract Number F33(657)-74-C-0588, Project No. 7381, Task No. 738107. Mr. Hal Dunegan, Dunegan/Endevco, served as Consultant. The effort described herein was conducted during the period 20 May 1974 to 20 November 1974 under the technical direction of Mr. W. H. Lewis. Mr. C. D. Bailey was the Principal Investigator and Mr. W. M. Pless performed the data analysis. The Air Force Project Monitor was Mr. Harold Howard from the C-5A Special Project Office. Mr. Thomas Cooper, AFML/MXA, and Mr. Grover Hardy, AFML/MXA, provided technical assistance to the Project Monitor. This report was submitted by the authors for publication March 1975.

Special recognition is due Messrs. Jim Tabb and Mike Roginsky for design and development of the system spectrum analyzer and to Mr. Sam Glass who performed the necessary tests during flight of the host aircraft.

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LIST OF ABBREVIATIONS

ALDCS	Active Lift Distribution Control System
AE	Acoustic Emission
D-A	Digital-to-Analog
db	Decibel
FSMS	Flight Structural Monitoring System
MHz, KHz	Megahertz, Kilohertz
PCM	Pulse Code Modulation
P-P	Peak-to-Peak
SCG	Stable or Slow Crack Growth
VCO	Voltage Controlled Oscillator
Structure-Borne Noise	Noise energy which is introduced into and propagates within the members of a structure
BL	Buttock Line
WL	Water Line
R	Right
L	Left

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INTRODUCTION

This Program was concerned with the measurement of structure-borne noise levels in a flying aircraft within sensitivity and frequency ranges sufficient to evaluate the feasibility of a flight acoustic emission (AE) system. The Program was conducted as a joint effort among the Lockheed-Georgia Company, the Dunegan/Endevco Company, and the Air Force Materials Laboratory. Lockheed managed the Program, assembled and flight tested the AE system, provided data analysis, and prepared the final report. Dunegan/Endevco provided transducer calibration and technical consultation services to support the Program. The Air Force provided the funds for special instrumentation build-up, instrumentation installation, data recording and travel. Management of the Program, analyses of data, and preparation of the final report were done with Lockheed internal research and development funds since the Program paralleled a continuing in-house AE program. The structure-borne noise measurements were made on C-5 Aircraft No. 0003 which was in flight test status to evaluate the performance of the Active Lift Distribution Control System (ALDCS). The acoustic emission structure-borne noise measurements program was conducted on a non-interference basis and none of the flight test costs and aircraft ground maintenance were charged against the program.

The purpose of this Program was to conduct precise noise measurements to determine whether a favorable signal-to-noise condition exists during aircraft flight for acoustic emission monitoring operations. The signal-to-noise ratio is considered to be the major limiting factor for the use of AE instrumentation and techniques on dynamic structures such as flying aircraft. For the success of this application, the voltage levels produced by structure-borne noise during flight must be significantly below the voltage level produced by a growing flaw AE signal.

An aircraft in flight literally resounds with noise which is generated principally by airflow, engines, and flight control systems. In the audible range this noise may be very high; but at ultrasonic frequencies, e.g., from 0.10 to 2.0 megahertz (MHz), where most AE systems are designed to operate, the noise is considerably lower, although it is often of sufficient intensity to interfere with AE signals. Acoustic emission signals, on the other hand, are low level at all frequencies, but may remain sufficiently high at ultrasonic frequencies to overcome the background noise.

Developments in AE instrumentation and techniques in recent years have greatly improved the potential for application to aircraft structures. Spatial and frequency discrimination techniques, differential transducers and minicomputers make possible the achievement of high signal-to-noise ratios and versatile signal processing.

Recent investigations conducted at the Lockheed-Georgia Company on complex structural specimens using such techniques have demonstrated the feasibility of detecting crack growth in a noisy environment (References 1 and 2). It remained to determine the feasibility of applying AE techniques for detection of stable crack growth in operating aircraft which have noise characteristics different from load-cycled specimens. One of the first steps in determining the feasibility is measurement of the aircraft's structure-borne noise over a frequency range compatible with AE instrumentation.

SUMMARY

This program was conducted to measure the structure-borne noise generated at certain locations on the airframe of a C-5A aircraft during flight. The purpose of the tests was to determine the feasibility, from a noise limiting standpoint, of using acoustic emission techniques to monitor slow crack growth on an aircraft. The primary measurement system used nine acoustic emission piezoelectric transducers and preamplifiers feeding into a sweep frequency spectrum analyzer. The output of the analyzer was directed to an on-board Pulse Code Modulation system for digitizing, then recorded on the aircraft's flight test instrumentation recorder. The nine-channel system was calibrated after installation on the aircraft by injecting a white noise signal into each channel at various signal levels and recording the output for later analysis.

Noise measurements were made on four flights at 10,000, 20,000, and 35,000 feet. The aircraft's time-code signal was used to correlate the noise measurements with flight events. Aircraft locations where transducers were installed included the center wing mid and rear beams, the inner wing front mid and rear beams in the pylon attach area, the fuselage/main landing gear area, and the empennage.

A second system using a commercially available acoustic emission instrument, the Dunegan Flaw Locator, was installed to monitor two areas on the center wing lower surface near the rear beam cap. This system uses transducer pairs to detect and provide information for locating growing cracks. The structure-borne flight noise sensed by this system was very low and no crack growth indication was obtained with the system during the four test flights. Thus, standard NDT methods for crack verification were not required.

Test results show that structure-borne noise will not prevent the use of an acoustic emission system designed to operate within the frequency range of 0.5 MHz to 1 MHz depending on the location-dependent noise background. Below 0.5 MHz the noise becomes too high to achieve a suitable signal-to-noise ratio (acoustic emission vs structure-borne noise). The highest noise was indicated in the main landing gear area and moderate noise was indicated in the pylon attachment area near the front beams of the inner wing. The center wing and the vertical stabilizer areas experienced below moderate noise levels. A suitable signal-to-noise ratio of 2:1 or better can be achieved for most of the moderate and low noise areas of an AE system operating at frequencies of 0.5 MHz or higher. The high noise area may require operation of up to 1.0 MHz to achieve the same signal-to-noise ratio.

DISCUSSION

1.0 OBJECTIVE

The purpose of this program is to demonstrate the feasibility of using Acoustic Emission (AE) techniques to detect stable crack growth in aircraft structure during flight by measuring the level of structure-borne flight noise within an AE-compatible frequency range.

2.0 APPROACH

A C-5 aircraft, LAC-0003, was in flight test status for the purpose of conducting other programs. Authorization to use the aircraft on a non-interference basis to measure the level of structure-borne noise during flight was an integral part of the contract under which this program was conducted. Piezoelectric acoustic emission sensors were acoustically coupled to the structure at 9 locations which included both high- and low-noise areas. Preamplifiers with 40 db fixed gain were installed near the sensors. The structure-borne noise was detected by the sensors and the amplified signals were transmitted to a specially designed spectrum analyzer for filtering and detection. The analog signals were routed through spare channels of the flight test Pulse Code Modulation (PCM) Data System (described in Reference 3) for conversion to digital form, then recorded on the flight test magnetic tape recorder. The data were later reduced to show voltage output versus frequency over the range from 0.1 to 2.0 MHz. To minimize costs and operate the AE system on a non-interference basis, existing FSMS electrical harnesses and instrumentation were used where practical, while maintaining an operational separation of the FSMS and AE/SCG systems.

In order to additionally demonstrate the feasibility of using AE techniques aboard a flying aircraft, two Dunegan Model 902 Flaw Locators were installed to monitor the structure-borne noise at two areas of the center wing lower surface near the rear beam inside the cargo bay. The material at all locations on LAC-0003 where transducers were installed is 7075-T6 aluminum alloy. From previous AE tests performed on aircraft specimens made of 7075-T6 aluminum and simulating structure geometry and configuration monitored in this Program, it is calculated that crack growth acoustic emissions produce a voltage level of at least 65 microvolts at the transducer terminals with typical test parameters and instrumentation.

2.1 Stable Crack Growth (SCG) Instrumentation and Installation

The entire AE/SCG system, including the flight test PCM components and recorder, is illustrated in block form in Figure 1. Hermetically sealed Dunegan/Endevco piezoelectric transducers, Model 9201, were the sensors used which were bond-coupled to the structure with an epoxy adhesive. At each location, a 3-foot cable connected the sensor to a 40 db-gain Dunegan/Endevco AE preamplifier, Model 801P, whose output is connected by a long signal transmission cable to the AE spectrum analyzer. The spectrum analyzer provides two analog and two digital signals to the Pulse Code

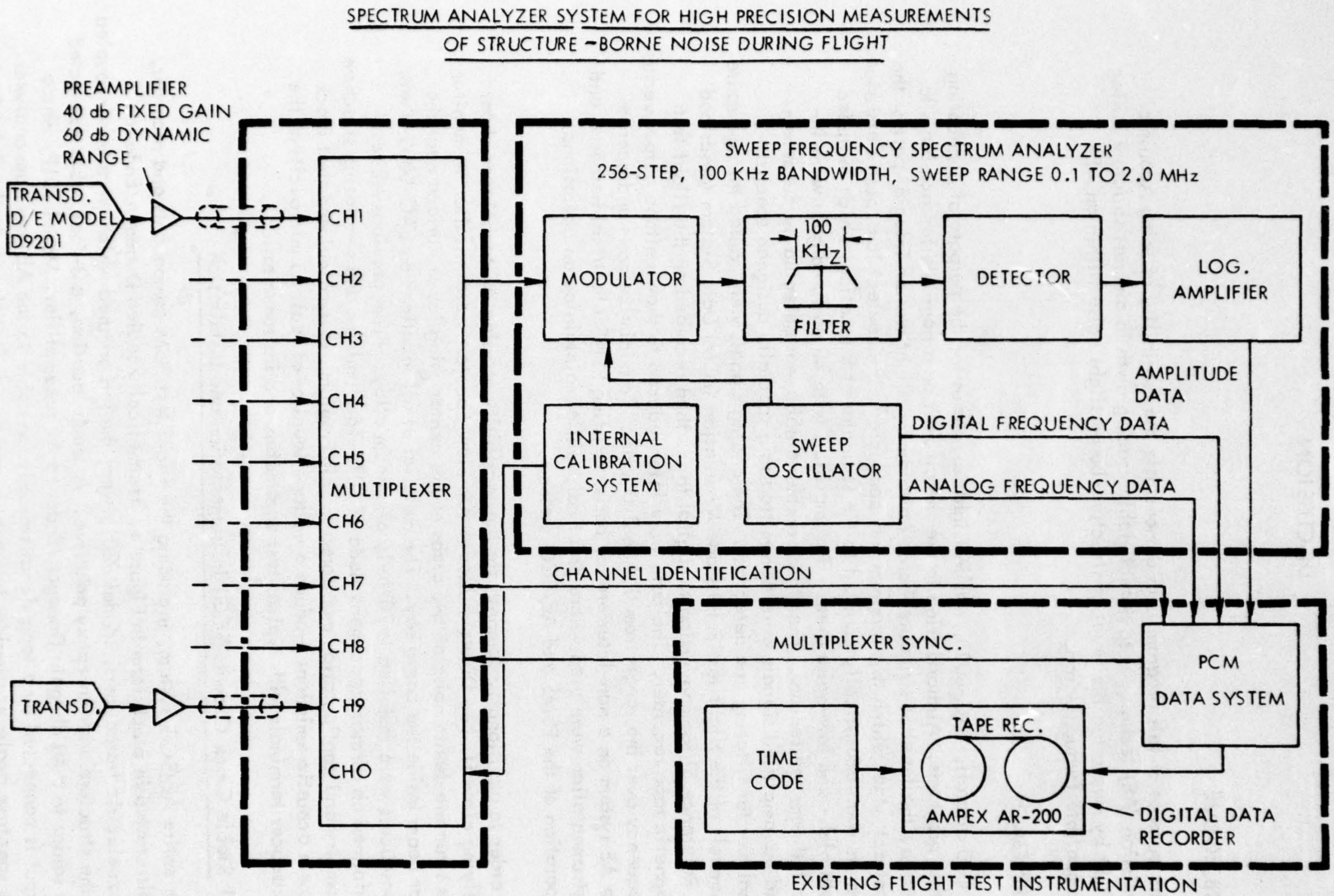


Figure 1. The Acoustic Emission Stable Crack Growth System Used to Measure Structure-Borne Noise During Flight.

Modulation (PCM) Data System. One analog channel carries the AE signal amplitude as a function of frequency and the second channel carries the ramp voltage whose instantaneous amplitude sweeps the spectrum analyzer and correlates to the signal frequency. The PCM converts the analog signal to digital form and transmits the digitized signal, multiplexed with the two digital channels, to the flight test magnetic tape recorder. The multiplexer channel number is encoded on one of the digital channels. The second digital channel contains an 8-bit binary number proportional to the analyzer's instantaneous center frequency.

The aircraft was instrumented at nine (9) locations with AE transducer-preamplifier combinations. These locations are shown on the sketches in Figure 2. The locations are numbered from 1 through 9 to designate the location and the corresponding transducer-channel input to the spectrum analyzer. The preamplifier and transducers are identified per serial number and location in the table in Appendix A. The locations are defined as follows:

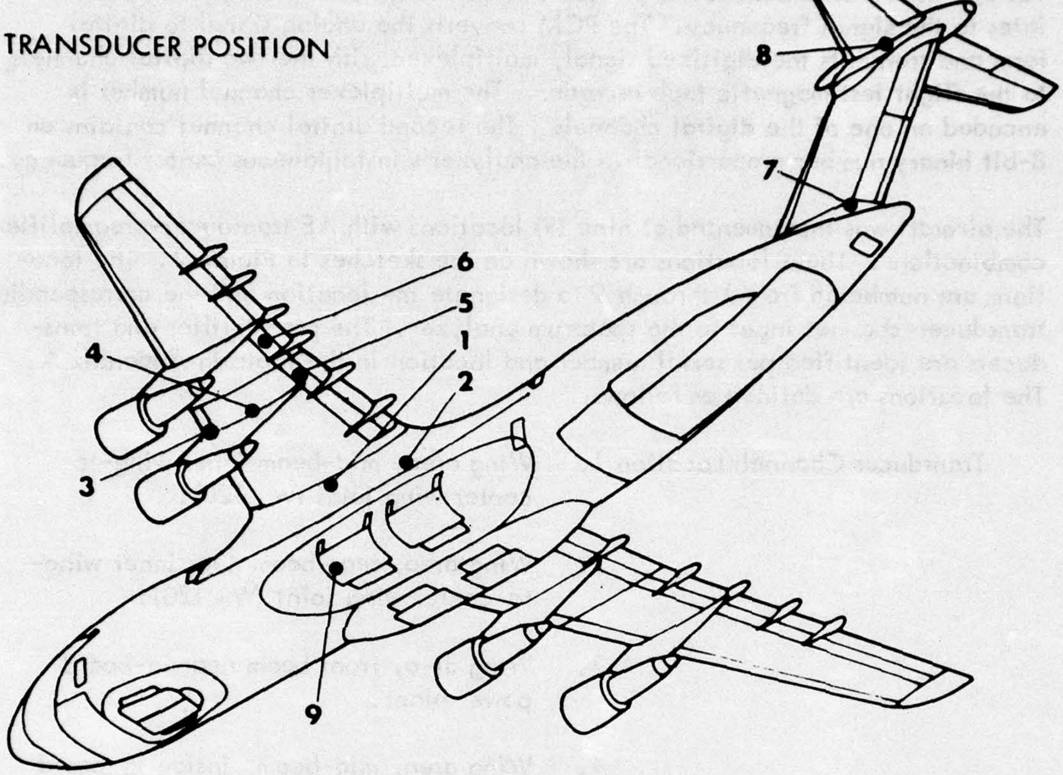
Transducer Channel/Location

1. Wing area, mid-beam inner wing-to center wing joint (WS 120).
2. Wing area, rear-beam near inner wing-to-center wing joint (WS 120).
3. Wing area, front beam near in-board power plant.
4. Wing area, mid-beam, inside in-board power plant nacelle.
5. Wing area, rear beam, behind in-board power plant.
6. Wing area, rear beam, inner wing-to-outer-wing joint (WS 577).
7. Empennage, vertical stabilizer-to-fuselage attachment.
8. Empennage, horizontal stabilizer near attachment to vertical stabilizer.
9. Main landing gear, structure near yoke attach.

Figure 3 consists of photographs showing all structure locations with the transducers and preamplifiers in place.

LOCATIONS FOR MONITORING NOISE LEVELS

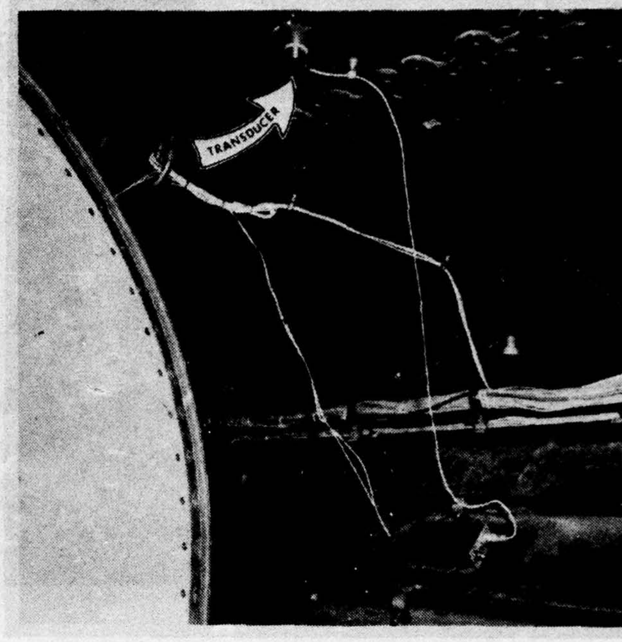
- TRANSDUCER POSITION



LOCATION	1	CENTER WING AT MID BEAM
	2	CENTER WING AT REAR BEAM
	3	INNER WING AT FRONT BEAM
	4	INNER WING AT MID BEAM AND INBOARD PYLON
	5	INNER WING AT REAR BEAM
	6	INNER WING AT REAR BEAM
	7	VERTICAL STABILIZER AT FUSELAGE
	8	VERTICAL STABILIZER AT HORIZONTAL STABILIZER
	9	FLOOR BEAM AT MLG ATTACH

Figure 2. Diagram Showing Locations of Transducer/Preamps on C-5A Test Aircraft.

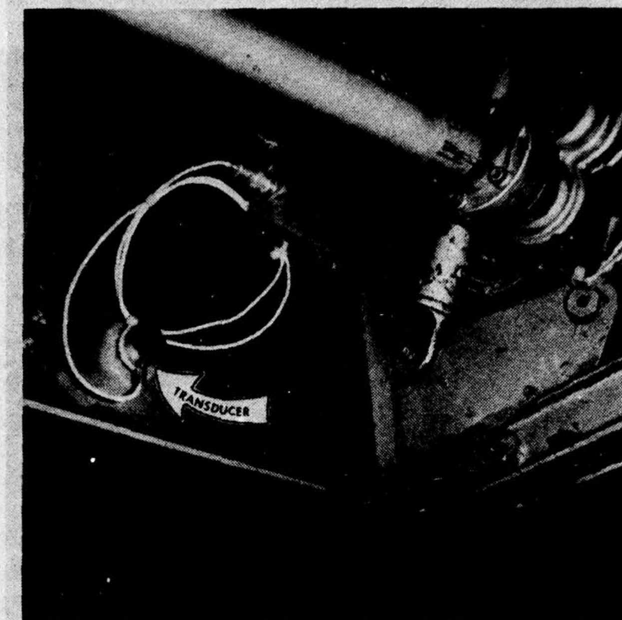
NO. 1 MID BEAM AT CENTER WING
TO INNER WING JOINT



NO. 2 REAR BEAM AT CENTER WING
TO INNER WING JOINT

Figure 3 (1 of 5). Photographs of Transducer/Preamplifier Installations on the C-5A Test Aircraft.

NO. 3 FRONT BEAM AT INBOARD PYLON



NO. 4 MID BEAM AT INBOARD PYLON

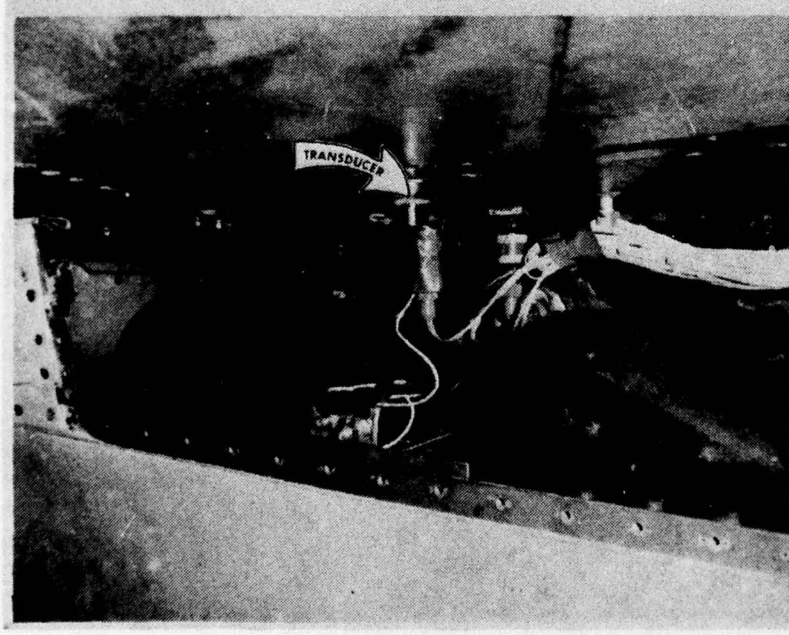
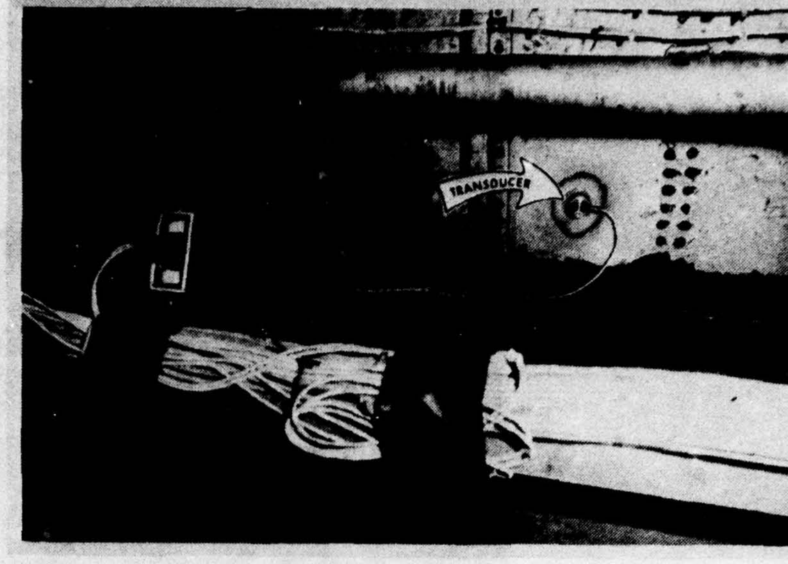


Figure 3 (2 of 5). Photographs of Transducer/Preamplifier Installations on the C-5A Test Aircraft.

NO. 5 REAR BEAM AT INBOARD PYLON



NO. 6 REAR BEAM AT INNER WING TO OUTER WING JOINT

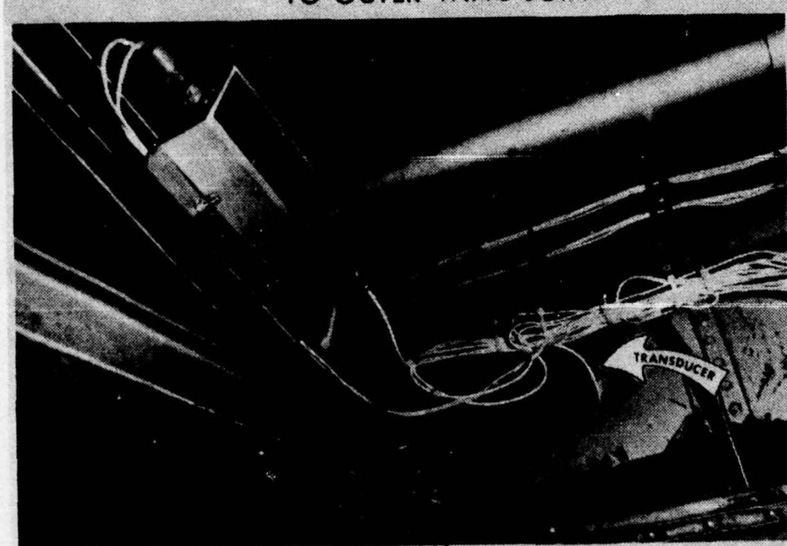
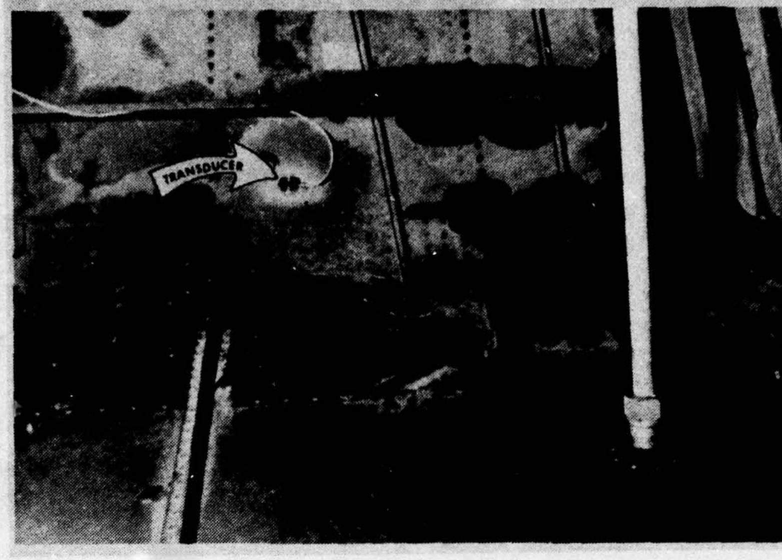


Figure 3 (3 of 5). Photographs of Transducer/Preamplifier Installations on the C-5A Test Aircraft.

NO. 7 VERITICAL STABILIZER NEAR ATTACHMENT TO FUSELAGE



NO. 8 HORIZONTAL STABILIZER

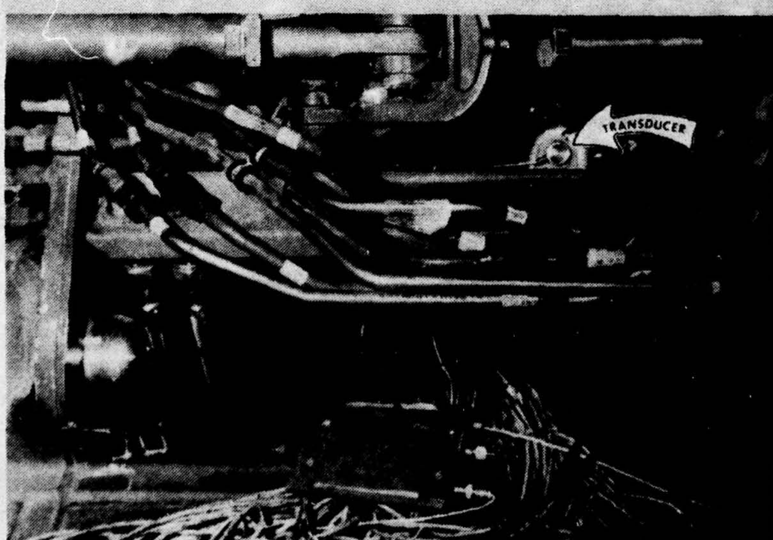


Figure 3 (4 of 5). Photographs of Transducer/Preamplifier Installations on the C-5A Test Aircraft.

NO. 9 STRUCTURAL MEMBER IN WHEEL WELL

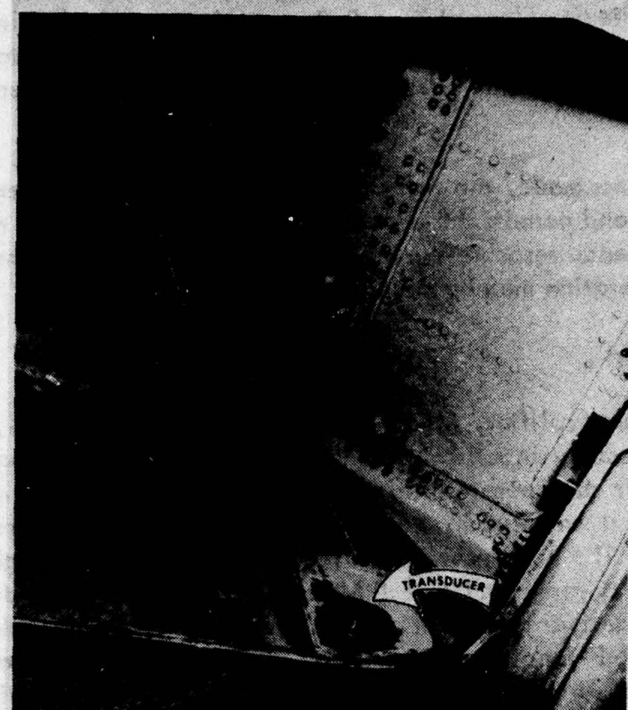


Figure 3 (5 of 5). Photographs of Transducer/Preamplifier Installations on the C-5A Test Aircraft.

2.1.1 SCG Transducers

The Dunegan/Endevco transducers, Model D9201, used in this Program and shown in Figure 4, are hermetically sealed units which have a flat temperature response between -100°F and +100°F and can operate up to +250°F with little variation. The sensor is electrically differential-ended, and is designed to respond to shear waves with high sensitivity. The sensitive element, a patented piezoelectric material designated as Piezite Element Type P-8, exhibits a nominal voltage sensitivity of -85 db referenced to 1 volt per microbar and a nominal capacitance of 235 picofarads. Frequency response calibration curves for the transducers used are in Appendix A.

Operating in the thickness mode, the transducer has a broad frequency response which is relatively flat and permits the unit to operate in the frequency range 0.1 to 2.0 MHz. The frequency response curves were obtained for each transducer using the spark bar calibration method developed by Dunegan/Endevco.

2.1.2 SCG Preamplifiers

The Dunegan/Endevco preamplifier, Model 801P, also shown in Figure 4, was used to amplify the transducer inputs prior to injection into the sweep spectrum analyzer. This amplifier can operate in either the single-ended or differential mode. In this Program, it was used in the differential mode for greater rejection of common mode noise. The 801P provides a fixed gain which varies slightly from 40 db across the frequency range 0.1 to 2.0 MHz. The amplifier has a dynamic range of 66 db and can deliver an output signal of ten volts peak-to-peak without distortion (10 millivolt P-P-input). The amplifier contains an integral bandpass filter having a roll-off of 18 db per octave. The +15 volt power is supplied through the output signal cable to minimize cabling needs. Spare cable harnesses which were installed in the wing and other areas for FSMS were used at all locations to avoid cabling redundancy and to minimize manpower costs. Prior to installation of these preamplifiers on the aircraft their responses were observed and measured at 71°F, 160°F, and -65°F, at several fixed frequencies between 0.1 and 2.0 MHz, using output impedances of 50 and 10,000 ohms. The response curves of each amplifier, i.e., output level versus frequency, are included in Appendix A, Figure A3.

2.1.3 Sweep Spectrum Analyzer

A specially-designed spectrum analyzer was built for this Program consisting of a channel multiplexer, amplifier and sweep frequency analyzer. It further provided a 4-channel input to the existing PCM data system in C-5A aircraft LAC 0003. The block diagram of the unit is shown in Figure 1 and Figure 5 is a photograph of the unit. An electrical schematic of the sweep frequency spectrum analyzer appears in Appendix B. Figure 6 is a photograph of the basic aircraft instrumentation installed in the cargo bay of LAC 0003 showing the locations of the analyzer and the PCM.

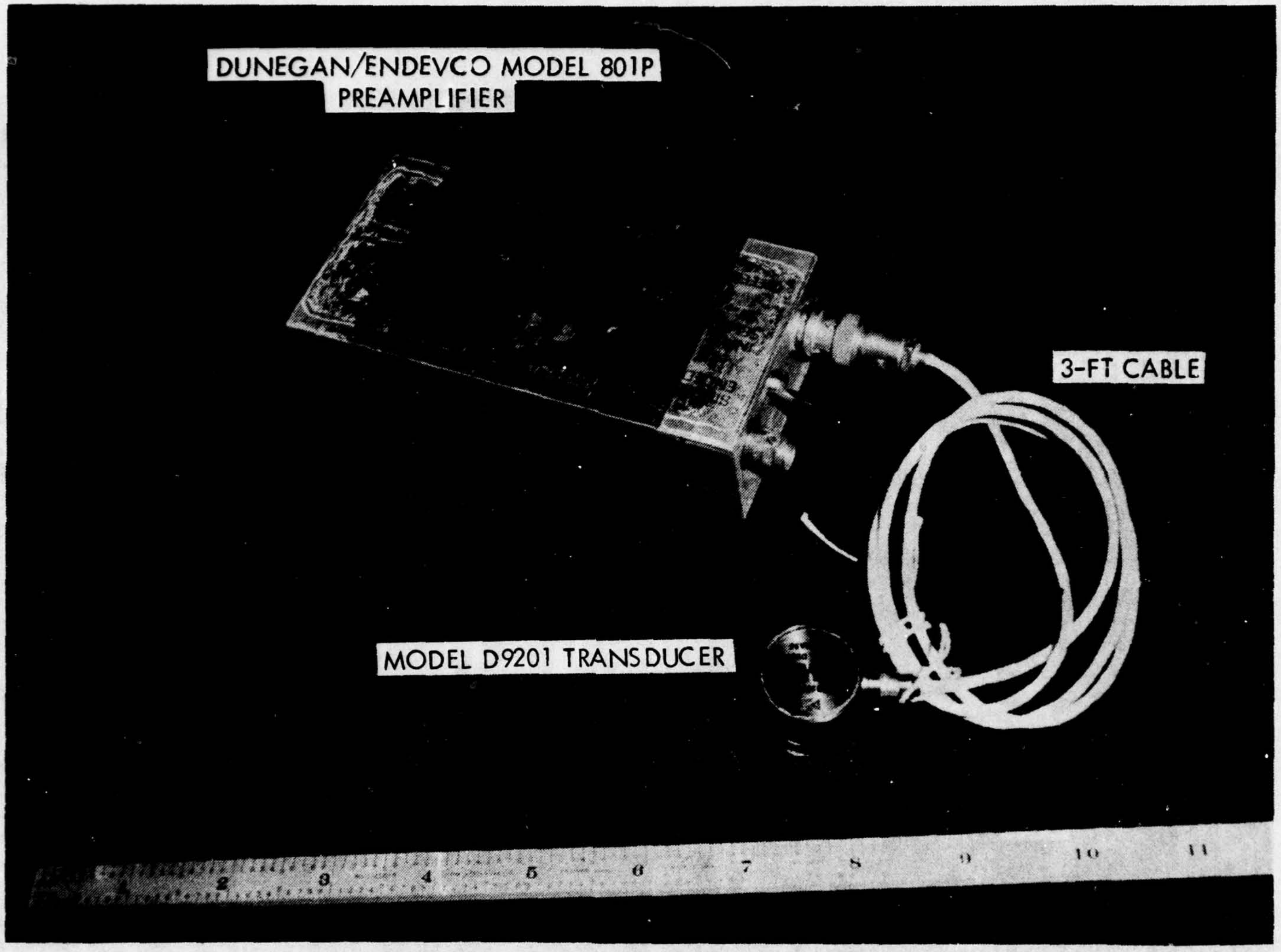


Figure 4. The Acoustic Emission Preamplifier and Transducer Used in the Measurement of Structure-Borne Flight Noise.

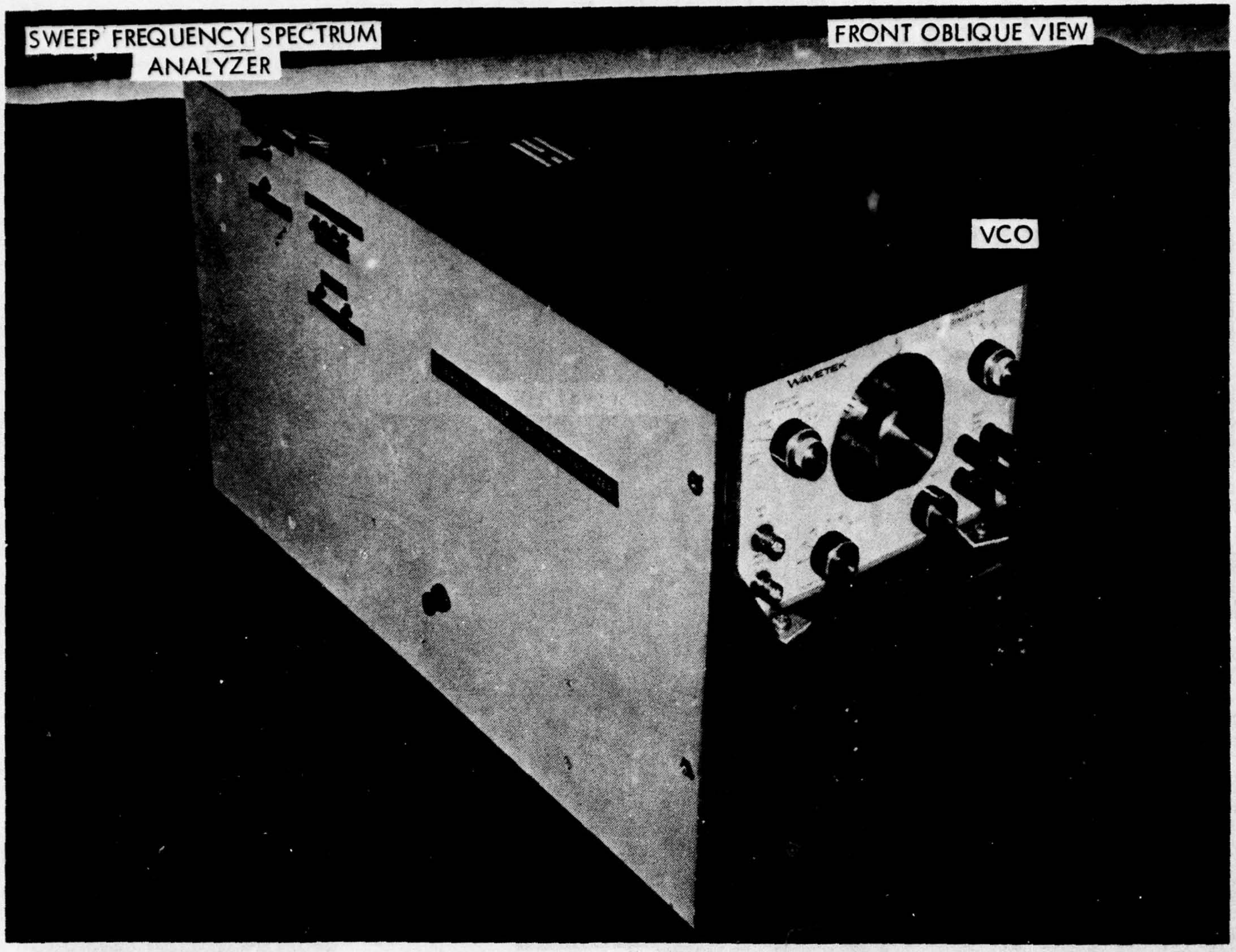


Figure 5 (1 of 3). Photograph of the AE/SCG Sweep Frequency Analyzer. Front View.

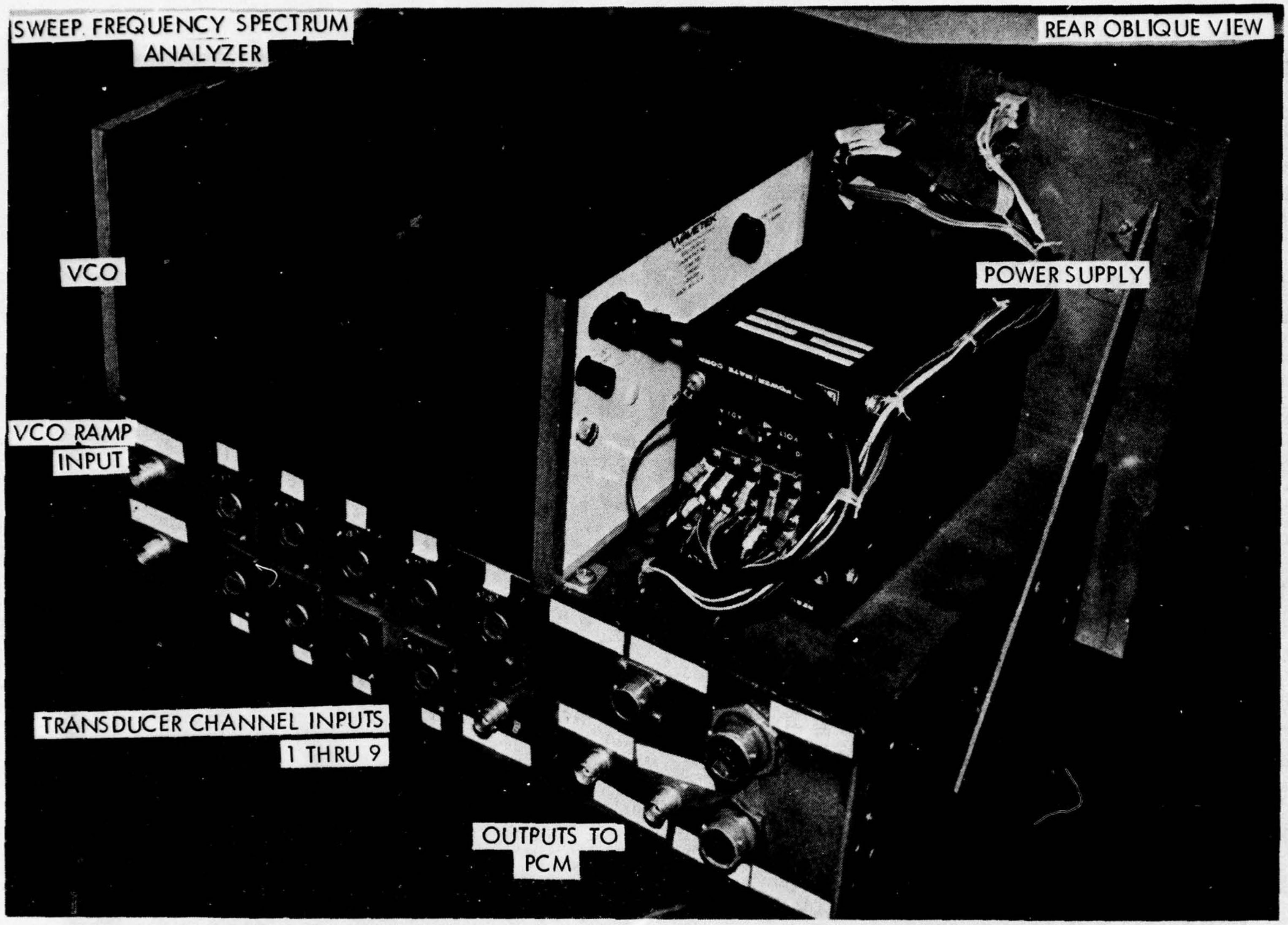


Figure 5 (2 of 3). Photograph of the AE/SCG Sweep Frequency Analyzer. Rear View.

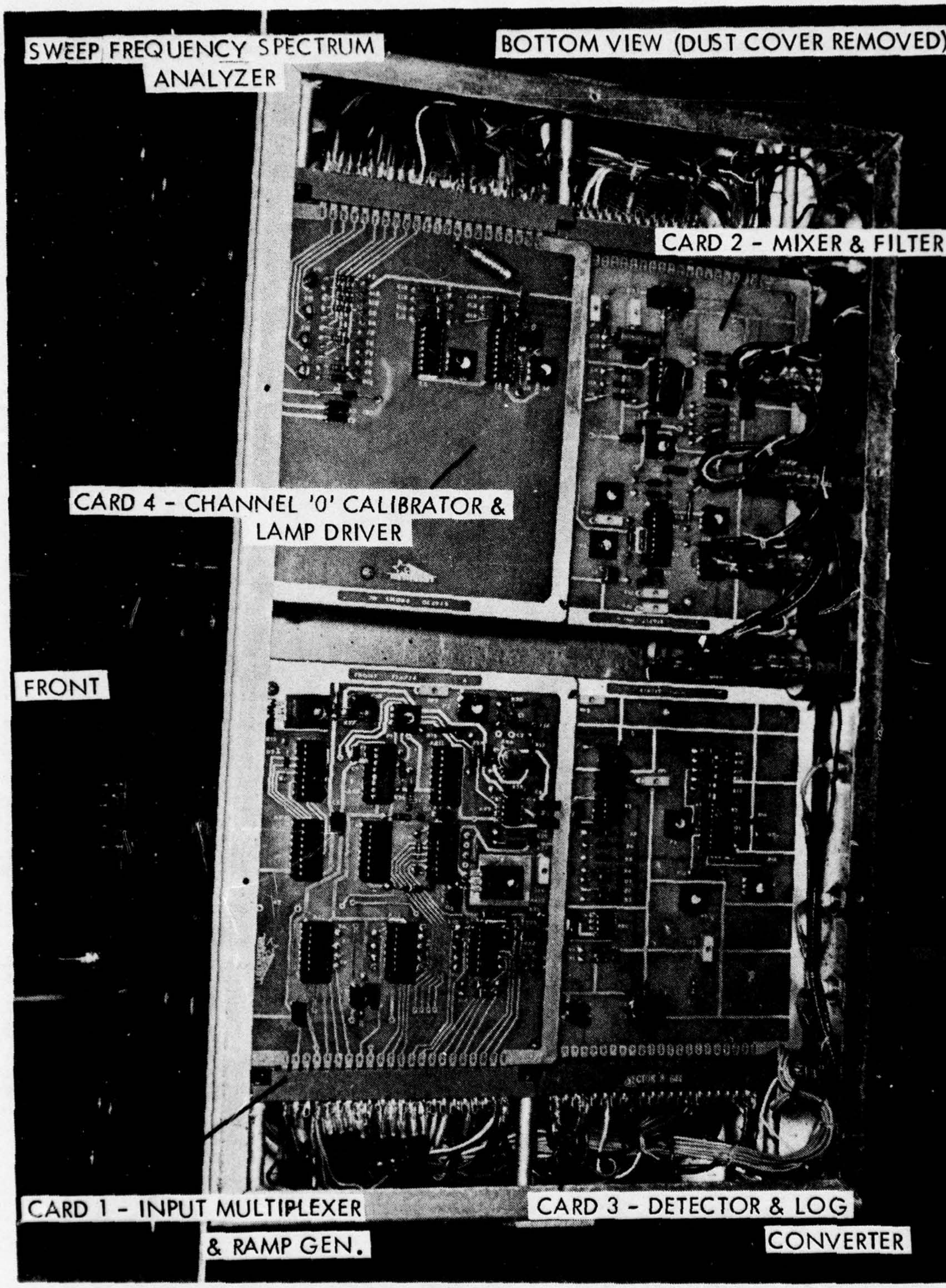


Figure 5 (3 of 3). Photograph of the AE/SCG Sweep Frequency Analyzer. Bottom View, Circuit Cards.

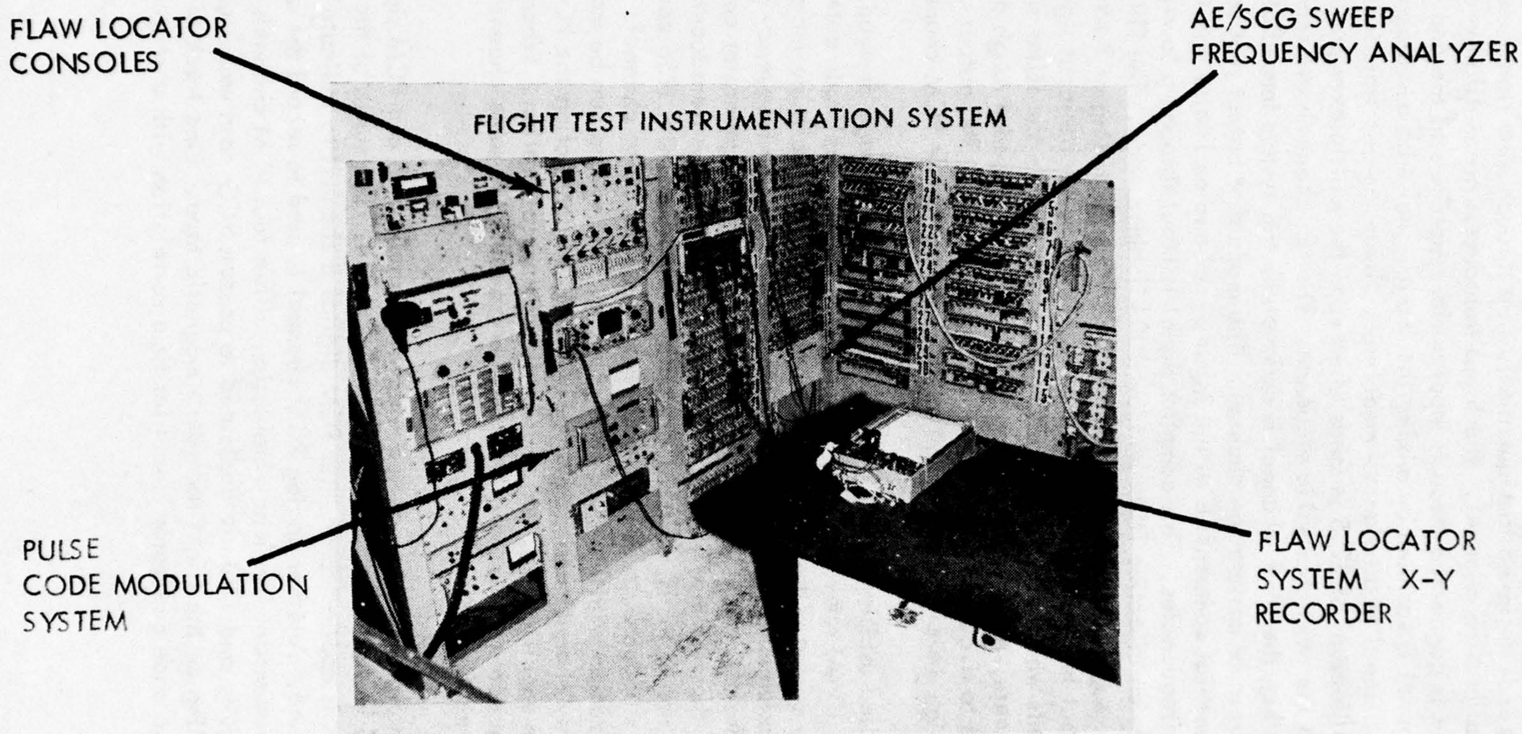


Figure 6. View of Equipment Bay Installed in the C-5A Test Aircraft Cargo Bay.

The spectrum analyzer is designed to sequentially step through nine transducer channels and one calibration channel. The input transducers are multiplexed so that one at a time is connected through input buffer amplifiers to the analyzer. Each transducer channel is scanned by mixing the input signal with a voltage controlled oscillator signal in a balanced modulator. Scanning one transducer channel requires an interval of 52.5 seconds whereupon the multiplexer proceeds immediately to scan the next channel in sequence. The ten channels are scanned in 8.75 minutes. When the ninth channel is completed, the system immediately returns to scan the zero or calibration channel. The modulator output products are the usual fundamental components of the input signal and oscillator and their sum and difference frequencies. This complex signal is input through a low-pass filter which provides an effective filter window 100 kilohertz wide. The filter, in effect, sweeps through a spectrum starting at 0.1 MHz and ending at 2 MHz. Thus, the filter output is a voltage proportional in amplitude to the input signal frequency components which are, at any given time, centered in the filter window. As the oscillator sweeps, the resulting output of the filter is passed through a detector, converted to d.c., and applied to the log converter. The output of the log converter is a high level analog signal which is routed to one PCM channel.

The voltage-controlled oscillator (VCO) is driven by a ramp voltage derived from a digital-to-analog (D-A) converter. The ramp instantaneous amplitude determines the frequency output of the VCO. The frequency ramp amplitude increases from minimum to maximum in 256 steps, which in effect scans the frequency band in 256 overlapping bands, each 100 KHz wide. When the ramp generator counter completes its ramp, the system will switch electronically to a new transducer input channel. The D-A converter is, in turn, driven by a binary counter (256 counts full scale) which is stepped by a 28-volt pulse from the PCM pulse channel. This maintains synchronization with the PCM frame timing. The system can be made to cease scanning at any channel and frequency by simply disconnecting the PCM channel pulse at the specific time, which is useful during calibration. When the pulse cable is reconnected, the system will resume scanning the same channel toward increasing frequencies.

The status of the ramp generator binary counter bits is applied to one PCM signal digital B channel to indicate the frequency being scanned. The status of the channel counter bits is applied to another PCM digital B channel to indicate the channel being scanned. Another analog PCM channel is used to record the actual ramp voltage for a redundant frequency indication. Thus four PCM channels (Channels 61, 62, 124, and 126) are dedicated to process SCG data which are available for recording on the flight recorder's magnetic tape. One track of the tape records the time code generator output for time correlation with observed events.

2.1.4 The Pulse Code Modulation* (PCM) Data System Flight Recorder, shown for reference in Figure 7

The PCM is essentially an analog-to-digital converter whose function is to prepare the data for recording and to synchronize the spectrum analyzer multiplexer. The PCM has a capacity of 135 data channels, four of which were dedicated to the AE/SCG project.

The PCM output goes directly to an Ampex AR-200 magnetic tape recorder normally used for recording flight events. The time generator is input into the recorder to provide time correlation with flight events. Five tracks of the tape capacity were used to record AE/SCG data, including the time code, during flight. The tape recorder has a 30 minute recording time for each roll of tape.

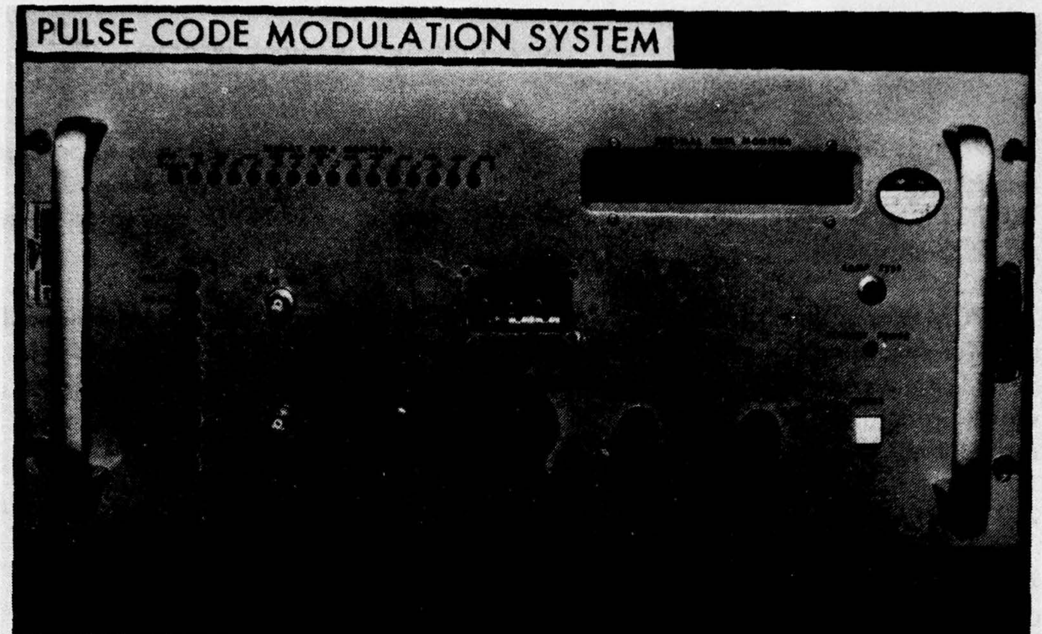
2.2 Flaw Locator System

A separate system from that described under Paragraph 2.1 was also installed on C-5A Aircraft LAC-0003 for flight tests. This system consisted of two Dunegan Flaw Locators, Model 902, their power supplies, four Dunegan Model 802PD preamplifiers, and 2 pairs of Dunegan Model D750 transducers, bond-coupled to the structure. The Flaw Locator system was operated at a center frequency of 750 KHz. A block diagram of the system is shown in Figure 8, and a photograph of the unit is in Figure 9. The Flaw Locators were installed in a single carriage in a 19-inch instrumentation rack within the aircraft cargo bay. An X-Y recorder, attached to a workbench in the instrumentation bay, was used to record the contents of the memory in the Flaw Locators at the end of each flight.

One pair of transducers was installed to monitor at BL 26L and the other pair was installed to monitor at BL 26R on the center wing box lower surface near the rear beam. The transducers had a separation of 36 inches and each transducer was connected to a preamplifier with a 3-foot long signal cable. The installed transducers were visible within the cargo bay as shown in the photograph of Figure 10. The amplifiers were connected to the Flaw Locator through 100-foot cables.

The two structural areas monitored with the Flaw Locator were selected in conjunction with the C-5A Stress Group. Criteria used in selection of these areas were: (1) they have a reasonable susceptibility to cracks, (2) they are accessible for installation of AE sensors, and (3) they are accessible for verification of any crack indications that may be obtained during monitoring. The purpose for using the Flaw Locator system was to assess the feasibility of using a readily available commercial AE system on board a flying aircraft.

*Model 371-81, Electro-Mechanical Research, Inc., Sarasota, Florida



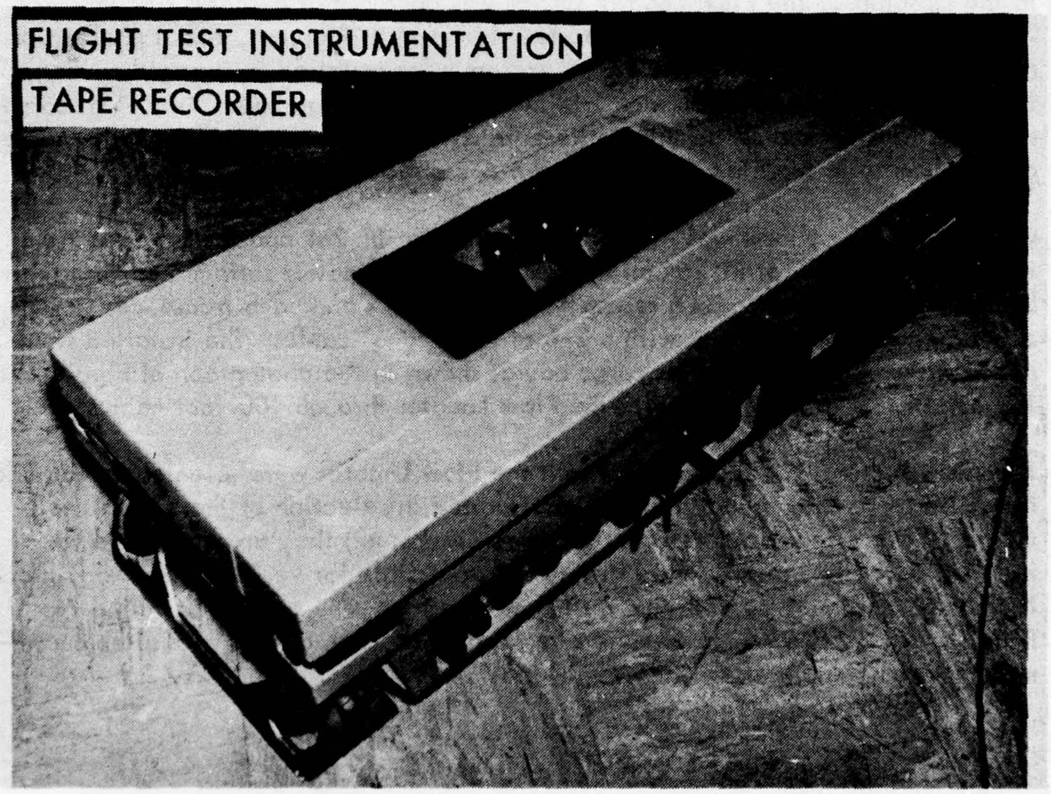


Figure 7. The FSMS Pulse Code Modulation Data System (Top) and the Flight Recorder (Bottom) Used with the AE/SCG System.

ACOUSTIC EMISSION FLAW LOCATOR INSTALLATION FOR FLIGHT TEST

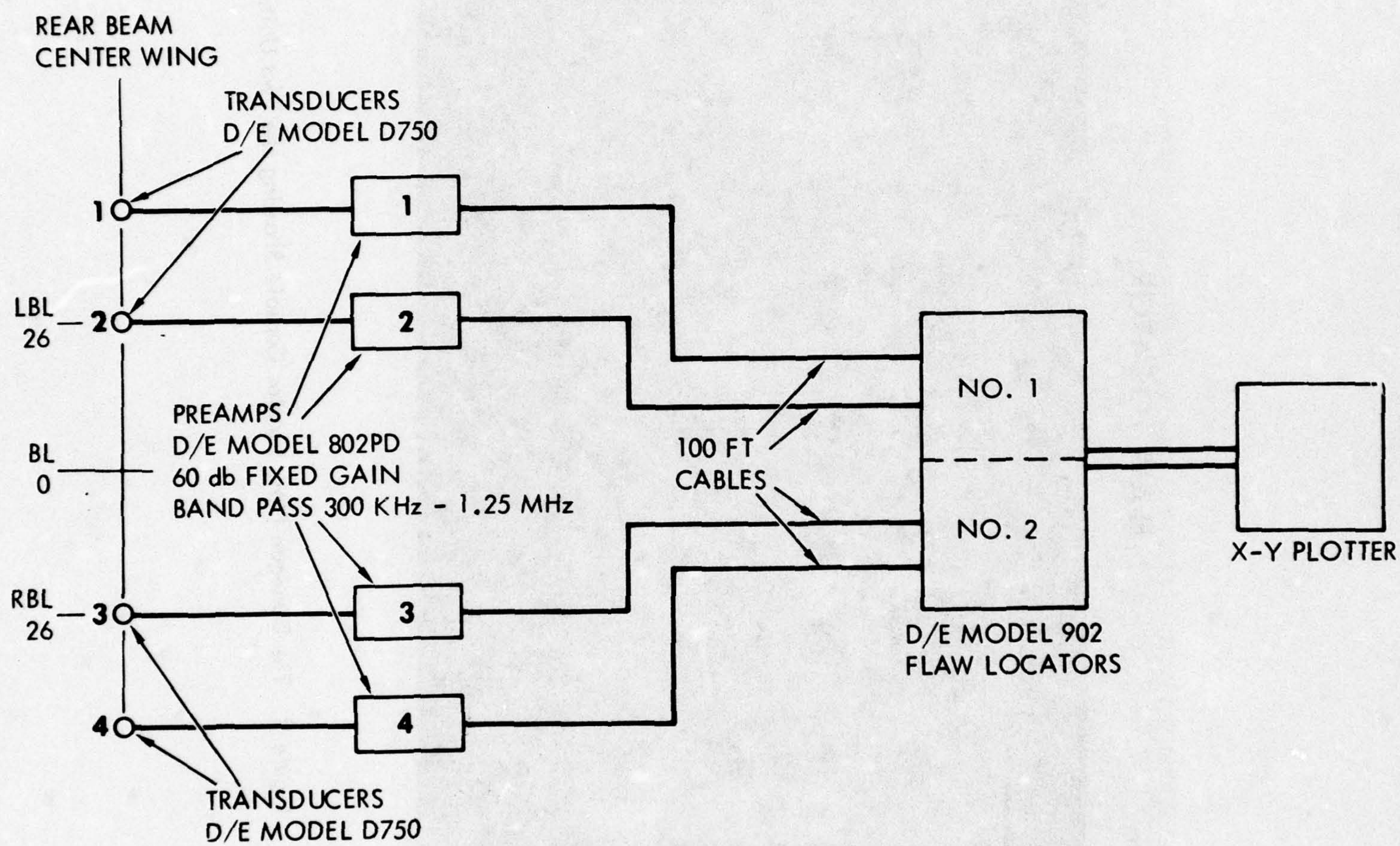


Figure 8. Diagram of the Dunegan Flaw Locator Acoustic Emission System.

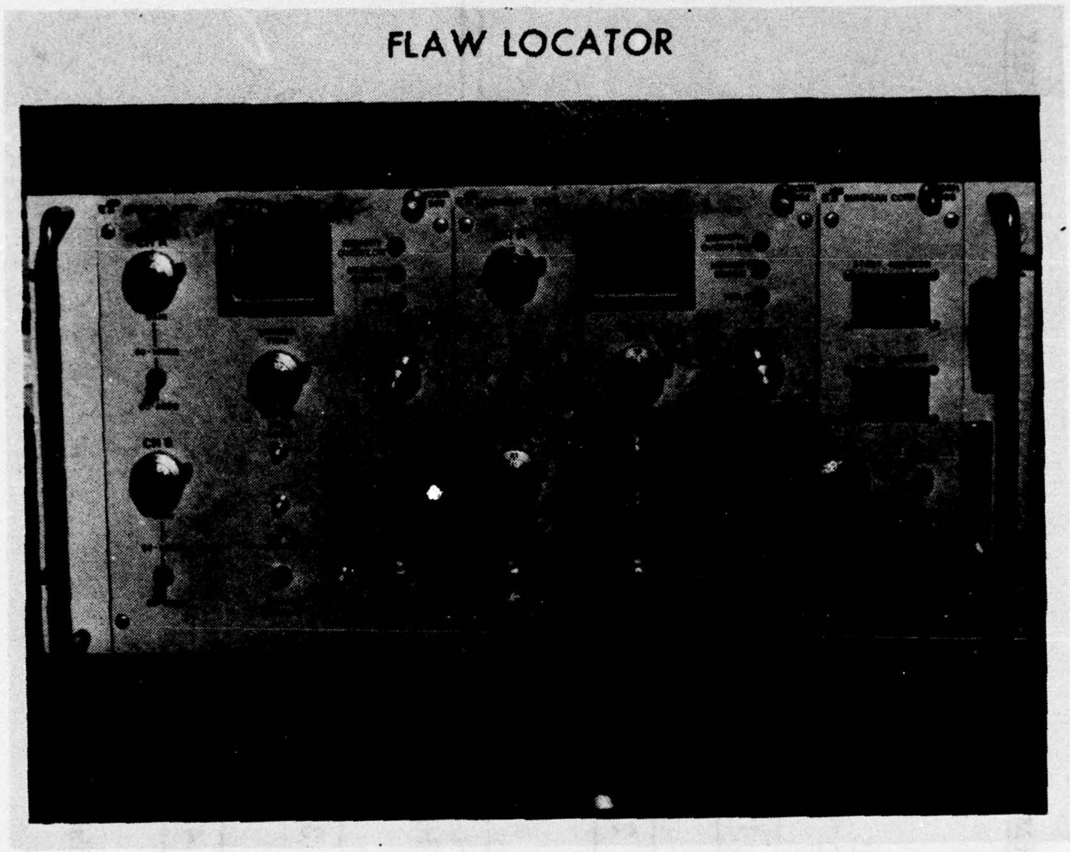


Figure 9. The Dunegan Flaw Locator Console Showing the Two Units

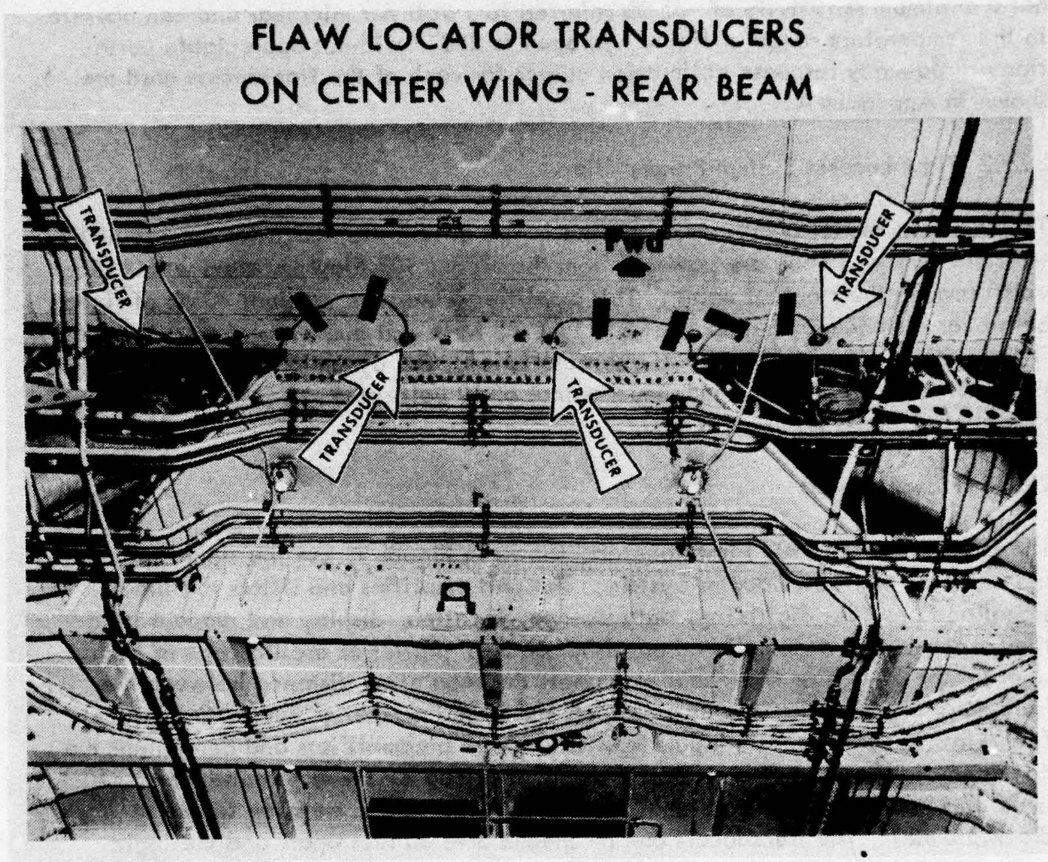


Figure 10. View Showing Installation of the Flaw Locator Transducers on the Center Wing Lower Surface.

2.2.1 Flaw Locator System Transducers.

The Dunegan Model D750 transducer is a differential sensor for high common mode rejection (> 40 db) of electrical noise (they were installed on the aircraft in the vicinity of hydraulic and air conditioning control systems). It is designed to operate in the thickness mode at a center frequency of 750 KHz. The sensor has a minimum sensitivity of -85 db referred to 1 volt per microbar and can operate in the temperature range of liquid hydrogen to 250°F without appreciable variation. Frequency response calibration curves for each of the transducers used are shown in Appendix A.

2.2.2 Flaw Locator System Preamplifiers

The Dunegan Model 802PD was operated in the differential mode for compatibility with the sensors. They are powered from the Model 902 Flaw Locators with +15 volts through their output cable. This amplifier provides a nominal 60 db of fixed gain over a frequency bandpass of 0.3 to 1.25 MHz. It contains integral bandpass filters with 18 db/octave roll-off characteristics. The dynamic range is 46 db and the amplifier can deliver a maximum output of 10 volts peak-to-peak with no distortion (10 millivolts input).

2.2.3 Flaw Locators

The Dunegan Model 902 Flaw Locators, shown in Figure 9, are the signal processing units for the Flaw Locator System. The unit amplifies and filters the input signal and provides digitizing, data storage, location, display and readout functions. Readout is accomplished on either an X-Y plotter or oscilloscope or both, wherein the X-axis or time-base represents the separation distance between the transducer pair and the Y-axis represents the stored number of events (amplitude). Each Flaw Locator accepts inputs from two paired preamplifiers and determines the difference in the time of arrival of a signal received by the two transducers, which is used to provide one-dimensional location of an acoustic emission source (e.g., crack growth). The transducers can be spaced 3 to 30 feet apart and the logic and controls in the Flaw Locator allow the operator to break the distance between the transducers into 1000 increments which becomes the limit of the spatial resolution for a given separation. This is accomplished because of the 1000 digital memory locations in the unit, each of which is capable of storing 256 events. The difference in time required for a signal emitted by a source between the transducers to reach each of the transducers is computed and assigned to the memory location corresponding to that time difference. When an event is stored, a three digit number is displayed on the front indicator giving the ratio of source distance to transducer separation distance. Repetitive emissions will cause a discreet buildup of counts in the corresponding memory location indicating a growing crack. Noise is also registered by the unit but usually occurs as events distributed more or less randomly through the 1000 locations. Occasionally, an active noise source will be displayed as a diffuse buildup of counts distributed over a number of closely spaced memory locations.

The Flaw Locator has a gain variable from 20 db to 60 db, so that the total gain when coupled with the Model 802PD preamplifier is 80 db to 120 db. The instrument was set for a total gain of about 86 db during the flight tests in this Program. The instrument effectively provides a spatial resolution of 0.036 inches for the 36 inch transducer pair separation. This resolution is greater than necessary for the particular application in our Program, but its effects are not objectionable.

The contents of the memory are continuously available on the oscilloscope terminals and can be provided on command (button switch) on the X-Y plotter terminals. Either presentation is a scan of the 1000 memory locations which also corresponds to the 1000 one-dimensional distance increments between transducers. Individual peaks that may occur on the X-axis are directly proportional to the locations and activity of the acoustic emission sources. Noise may appear as distributed "grass" along the X-axis with no well-defined peaks. In this Program an X-Y plotter was used to record and display the Flaw Locator's memory contents of the end of each flight and prior to power termination.

3.0 SYSTEM CALIBRATION

3.1 AE/SCG System

In order to establish operating parameters and determine the system response, the AE system was calibrated after installation on the aircraft. A compatible white noise source was used to measure system dynamic range, spectral response, and channel crosstalk. The test equipment arrangement for these tests is shown in Figure 11. In addition, a pulse transmitter was placed on the structure in proximity to each transducer to determine channel response to a calibration pulse. The test equipment arrangement for this test is shown in Figure 12.

3.1.1 System Dynamic Range

The random noise generator was connected through a precision 0-80 db attenuator to the differential input of the 801 preamplifier (refer to Figure 11). The preamplifier is connected to Channel 1 of the spectrum analyzer through a short (< 10 feet) cable which is also connected to the PCM and flight recorder. The channel sweep is allowed to scan through Channel 0 and begin the Channel 1 sweep. After 12 seconds, the PCM channel pulse cable is disconnected so that the analyzer stops its sweep and is locked at 500 KHz. The PCM is set to display the analyzer amplitude channel on PCM Channel 61. With the system locked on Channel 1 at 500 KHz, the attenuator was set to 20 db and the noise generator adjusted for $\pm$1800 counts on the display. The attenuator was adjusted through its range to assure proper operation of channel display, then reset to 20 db.

At this point the recorder was started, the system remained locked at 500 KHz, and the attenuator was switched through this sequence in two-second steps: 20, 10, 0, 10, 20, 30, 40, 50, 60, 70 and 80 db, then the recorder was stopped. This

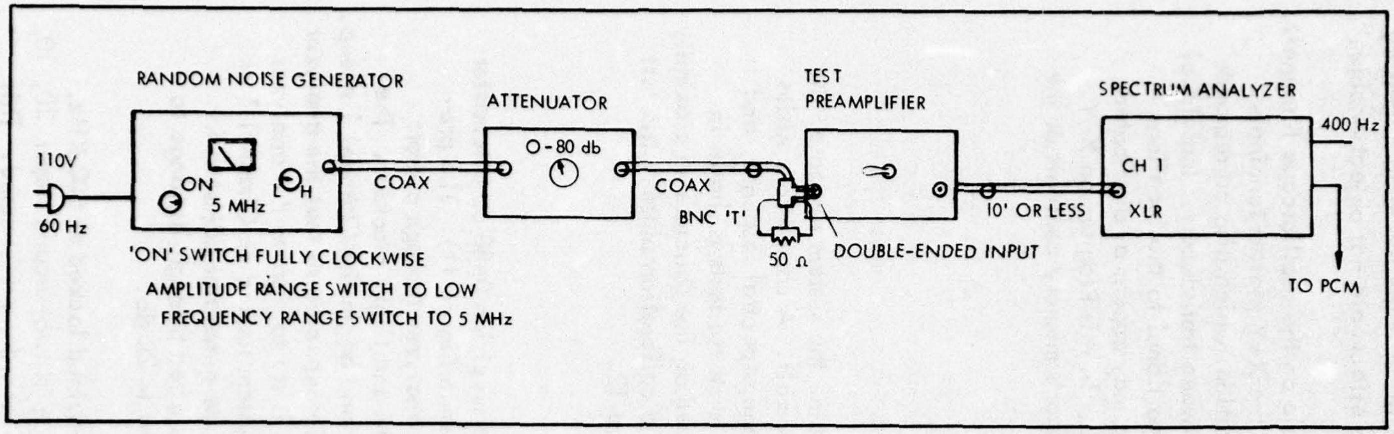


Figure 11. The AE/SCG Preflight Calibration Test Arrangement.

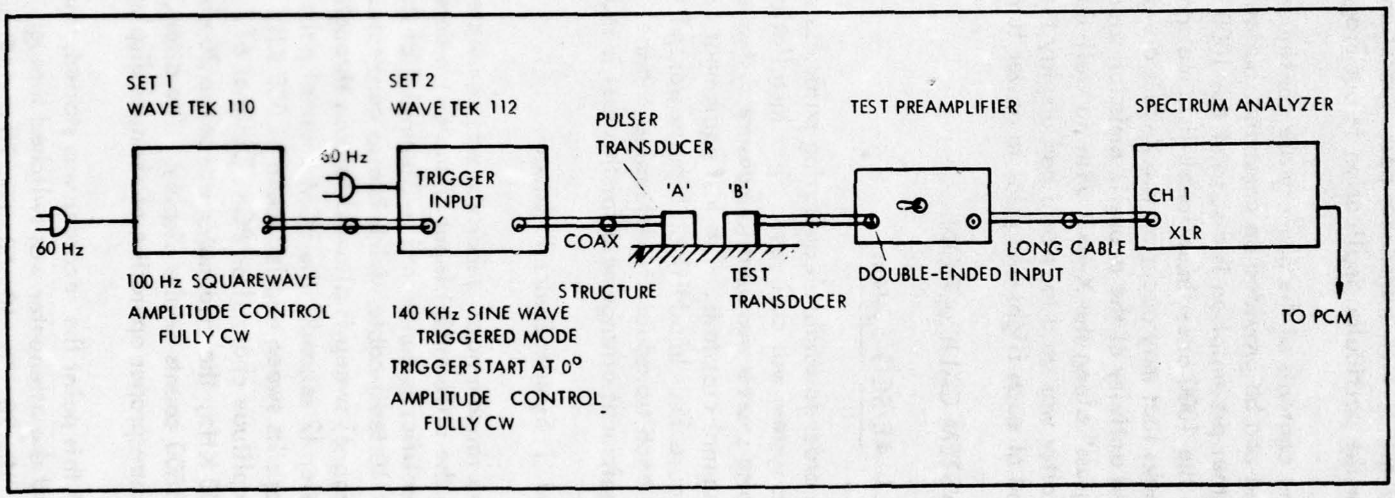


Figure 12. The AE/SCG Calibration Pulse Test Arrangement.

provided the dynamic range check of the system and is illustrated in Figure 13. The dynamic range of the system is greater than 60 db.

3.1.2 Baseline Frequency Response, Calibration Channel, and Crosstalk Checks

To obtain these checks, the test system described in Figure 11 was used. The attenuator was set to 20 db and the Channel 0 lock switch set to Channel 0. The channel pulse cable from the PCM was reconnected and the tape recorder started. After 60 seconds, the lock switch was set to normal and the analyzer scanned through Channels 1 and 2 (52.5 seconds per channel) while recording. The Channel 9 preamplifier was connected through Channel 1 for the initial tests using a short cable less than 10 feet long. The recorded information on Channel 1 then contains the combined preamplifier response and spectral noise source output variations. This response is shown as Curve A in Figure 14 and also in Appendix C. The noise generator spectral output was obtained by subtracting the No. 9 preamplifier gain from the system baseline response at specific frequencies and is shown as Curve C in Figure 14. Channel 0 contains the Calibration Channel check in which the fundamental peak near 200 KHz represents two volts P-P referred to the spectral analyzer input (the other peaks are odd harmonics, contained in the square wave whose relative amplitudes are shown in Figure C2 of Appendix C). The difference between the responses of Channels 1 and 2 provides the channel crosstalk check. The results show that crosstalk between channels is 25 decibels separation or greater for all channels.

3.1.3 Cable Attenuation and Frequency Response Checks

After making the baseline checks above, the test equipment settings were verified and locked so that the noise generator and attenuator could be moved to transducer locations as installed on the aircraft. This equipment was connected to the input of the preamplifier at remote installation locations and a complete frequency sweep was made within each channel. The equipment was connected first to the preamp/transducer location 9 for initial calibration, then to preamp/transducer locations 1 through 8 in sequence and the respective response curves recorded. These curves include the long cable attenuation and frequency response of each channel referred to the analyzer input as shown in Figure 14 (Curve B for Channel 9) and Figure 15 (Channels 1-8). These curves indicate the response characteristics the system will produce for a given signal output at the transducer.

3.1.4 System Pulse Response

The test equipment was set up according to the arrangement in Figure 12 to evaluate the response of each transducer channel (B_n) to a pulse introduced in the nearby structure by a 140 KHz pulsed transducer (A)*. The purpose of this test was to

*Dunegan/Endevco Model S-140 which has a center frequency of 140 KHz.

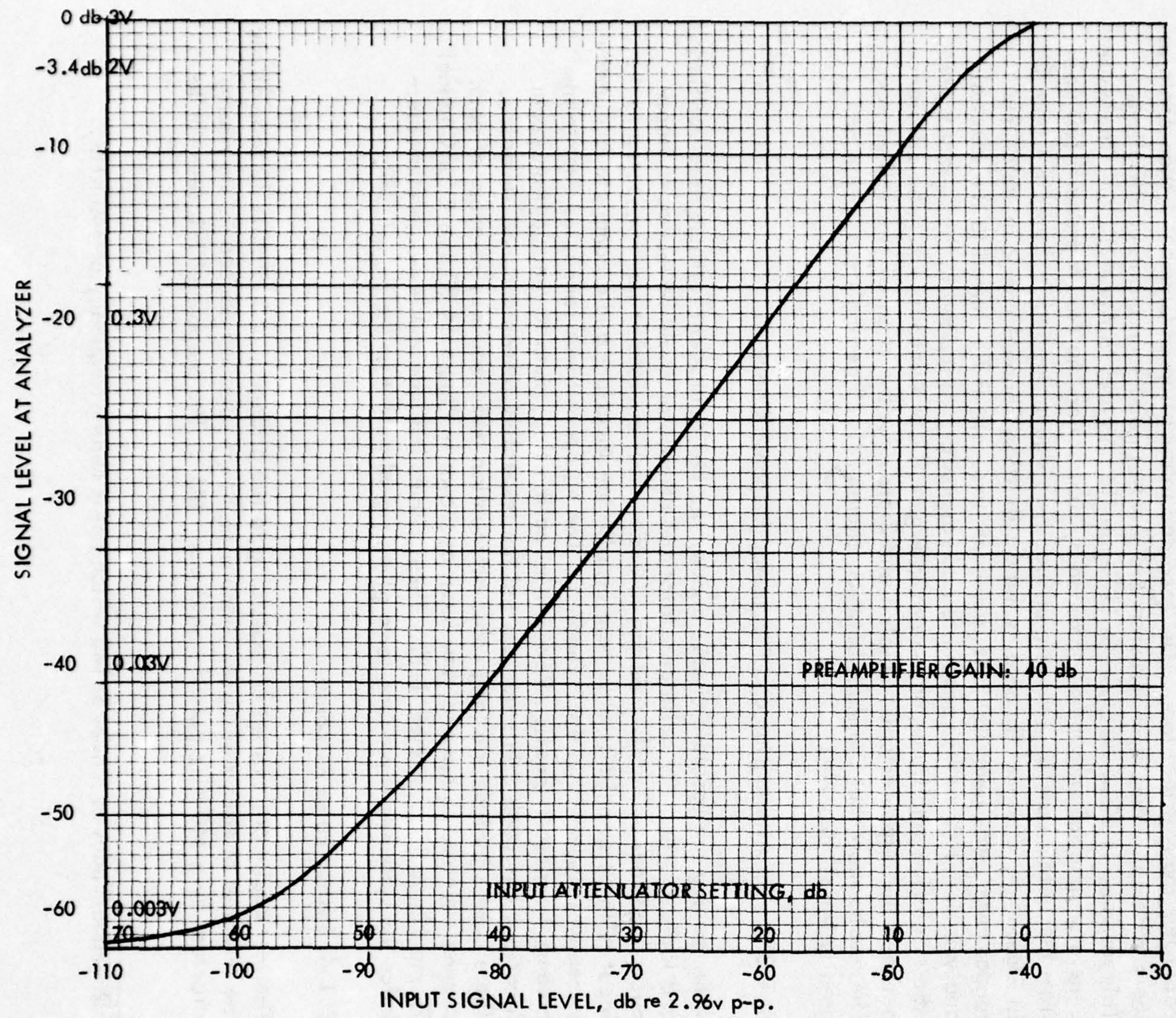


Figure 13. AE/SCG System Dynamic Range

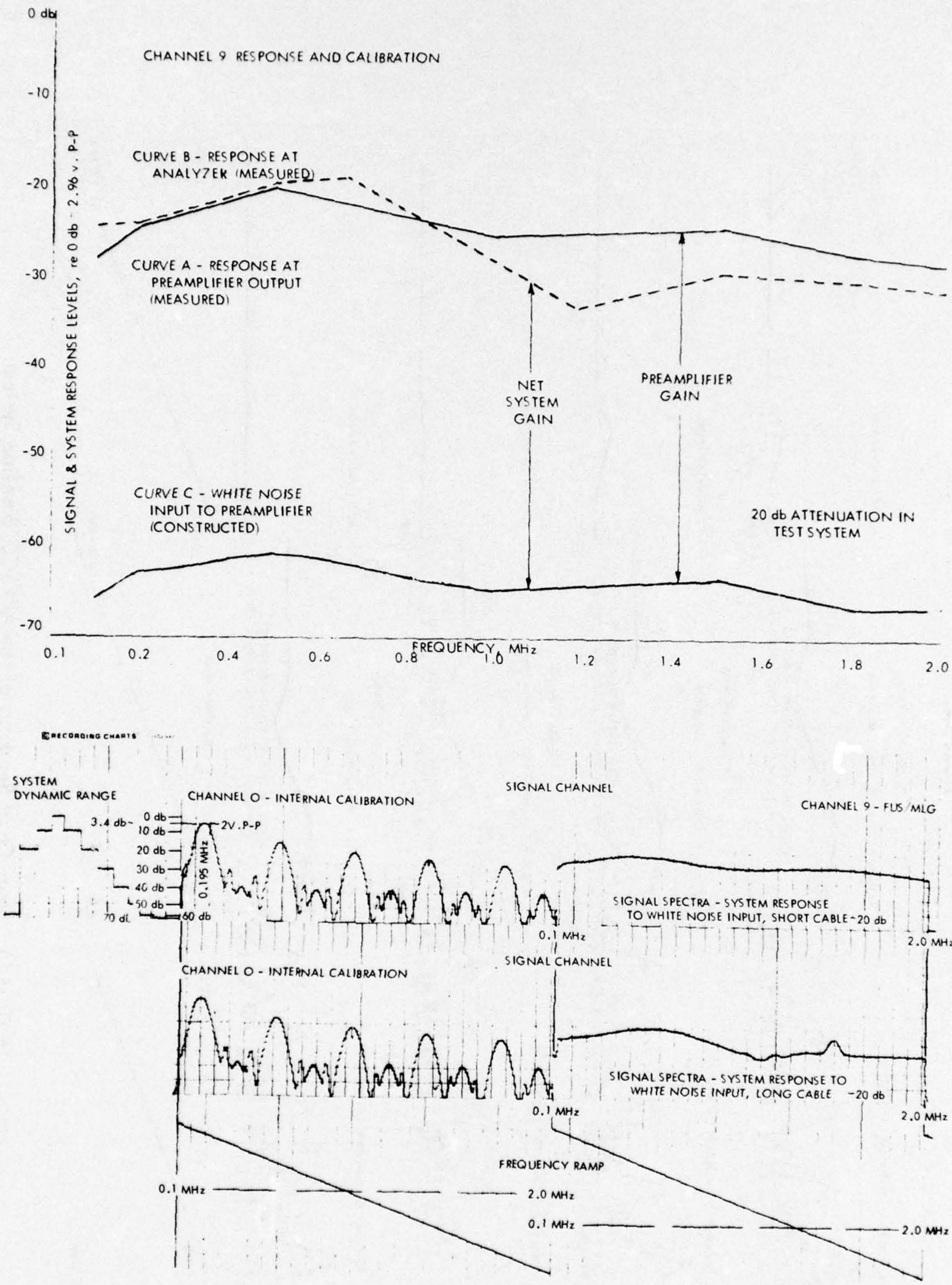


Figure 14. The Channel 9 Baseline System Response (Calibration). Bottom View Shows the Strip-Chart Tape Readout of the Preflight Calibration Tests. Topview Shows Analysis to Obtain System Gain and Noise Input Levels.

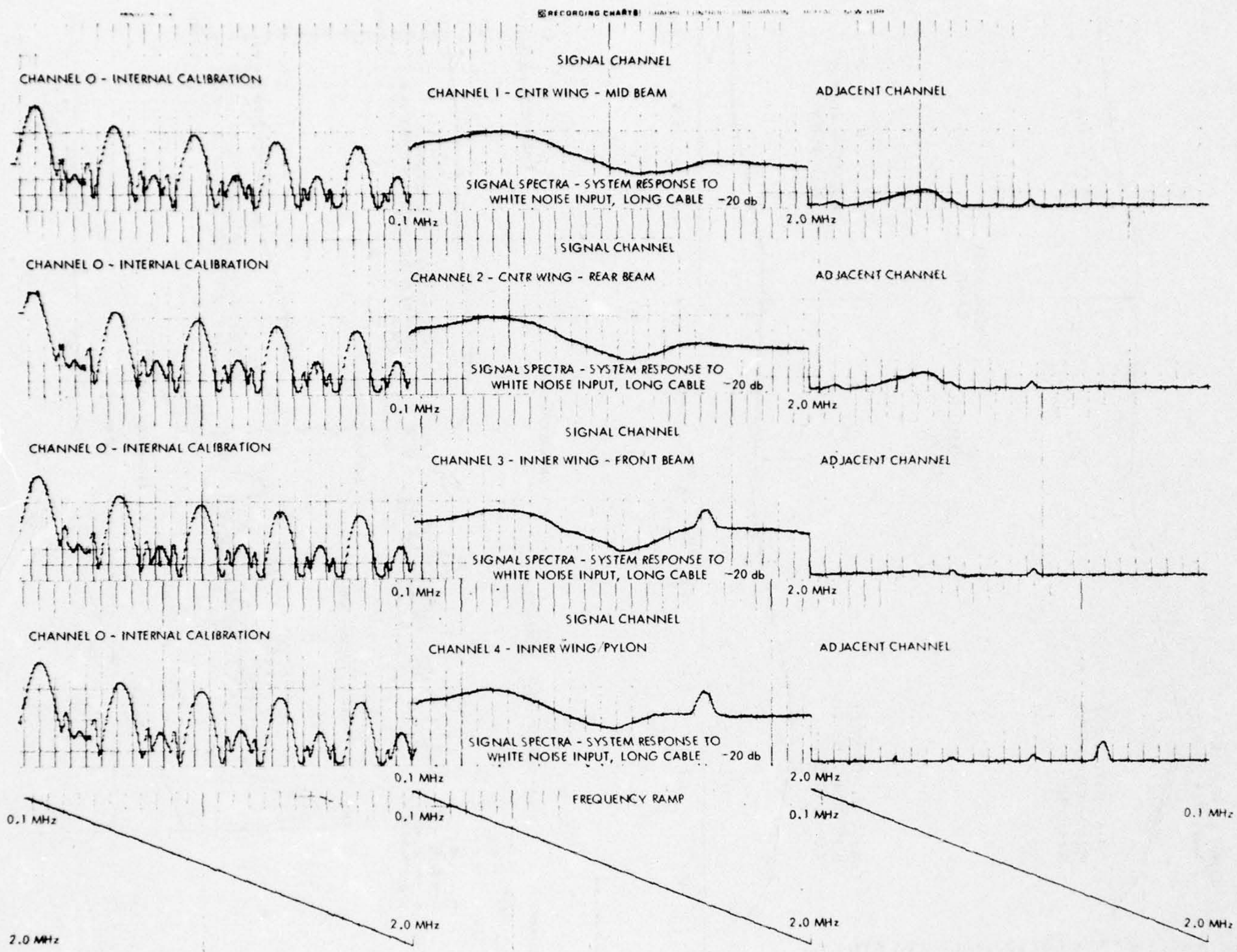


Figure 15 (1 of 2). Strip-Chart Readouts of the AE/SCG Baseline System Response for Channels 1 through 8 (1-4 this Page).

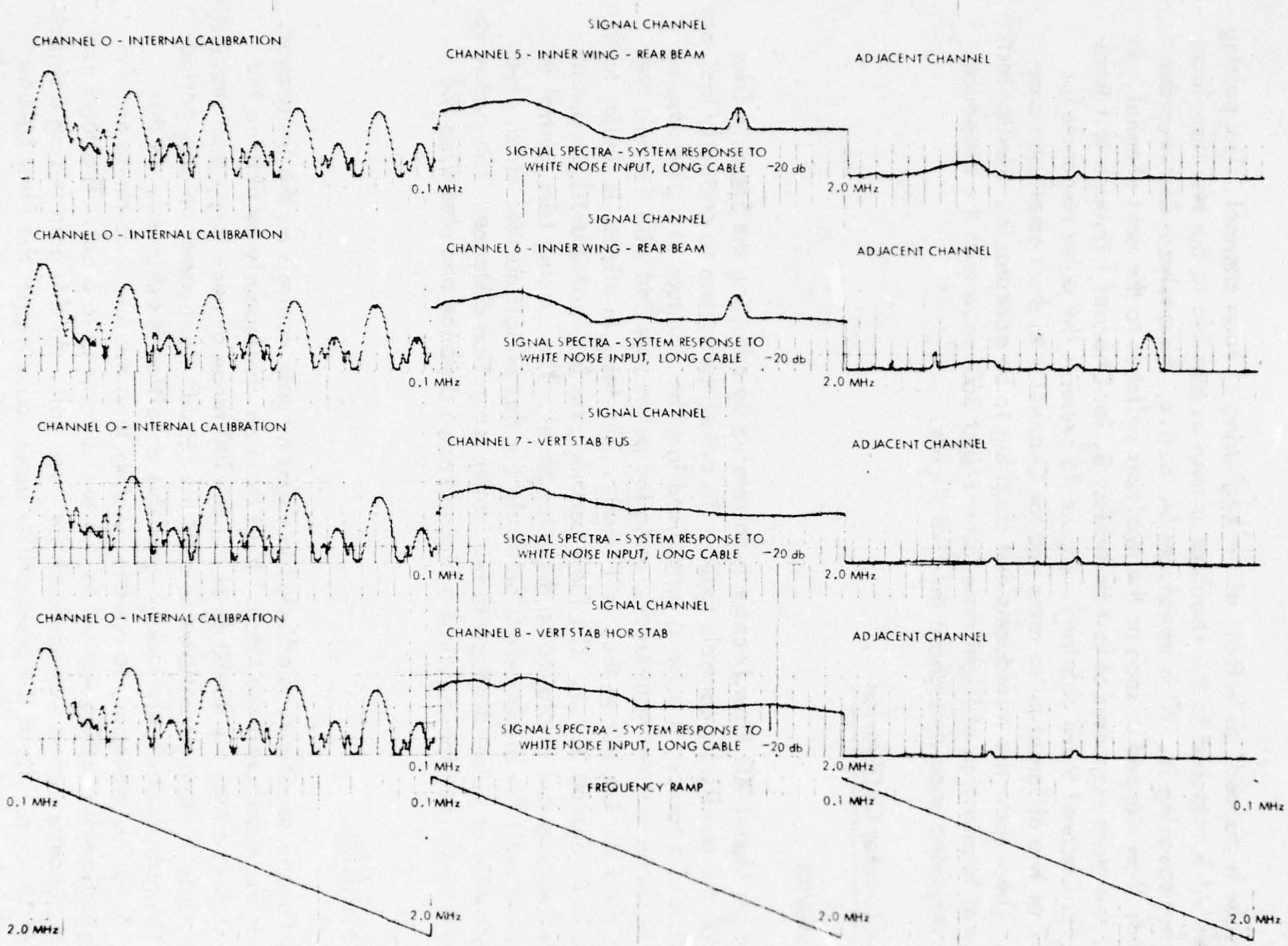


Figure 15 (2 of 2). Strip-Chart Tape Readouts for the AE/SCG Baseline System Response for Channels 1 through 8 (4-8 Shown this Page).

check the integrity of the transducer-to-structure coupling and the gross sensitivity of the transducer. Poor coupling or an "insensitive" or "dead" transducer would be clearly indicated in the response.

The analyzer is stopped, as before, at the beginning of each channel. The pulsing transducer 'A' is attached to the structure as near as possible to the test transducer 'B_n' without touching it. With transducer 'A' active, the analyzer and recorder are started, then stopped as soon as the analyzer switches to the next channel. In this way, the pulse responses of test transducer B_1 for Channel 1 through test transducer B_9 for Channel 9 are obtained. Figure 16 presents the pulse responses for Channel 7 at two different pulse rates and for Channel 1 to show amplitude comparisons. These show the transducers and coupling to be acceptable. Similar tests performed at termination of flight tests (after Flight 360) show that the transducer integrity remained good throughout the five flights.

3.2 Flaw Locator Calibration

3.2.1 Spacing

The Dunegan Model 902 Flaw Locator is calibrated by adjusting the Spacing Trim control and the sensitivity controls (gain). To adjust the system to correctly indicate the location of a source, a pulse is introduced into the structure at a premeasured distance between the two transducers, using the pulser supplied with the unit and operating the Flaw Locator in the "Test" mode. The test, in effect, is similar to that described in Paragraph 3.1.4. The Flaw Locator gain is adjusted until consistent and stable readings are displayed on the front panel. The spacing trim control is then adjusted until the display reads precisely the three-digit number which gives the correct ratio of pulser distance to transducer separation distance. The system can be checked by placing the pulser at a second known distance and observing the readout.

3.2.2 Sensitivity

The Flaw Locator sensitivity is adjusted in flight by gain controls on the instrument. The gain is increased until the "Hold" light stays on continuously by placing the toggle switch in either the 60-80 db or the 80-100 range as necessary, then readjusting the appropriate 10-turn potentiometer until the "Hold" light comes on only periodically. The gain controls are locked when this condition is achieved. In this Program, it was not necessary to operate in other than the 60-80 db range and the 10-run potentiometer setting was set to 2.5 to 3.0 turns (5 to 6 db). The total gain of the Flaw Locator system was 85-86 db for each unit. At this gain setting a signal as small as 50 microvolts at the preamplifier input could trigger the Flaw Locator counting circuits. This is sufficient to detect acoustic emission events from stable crack growth should they occur.

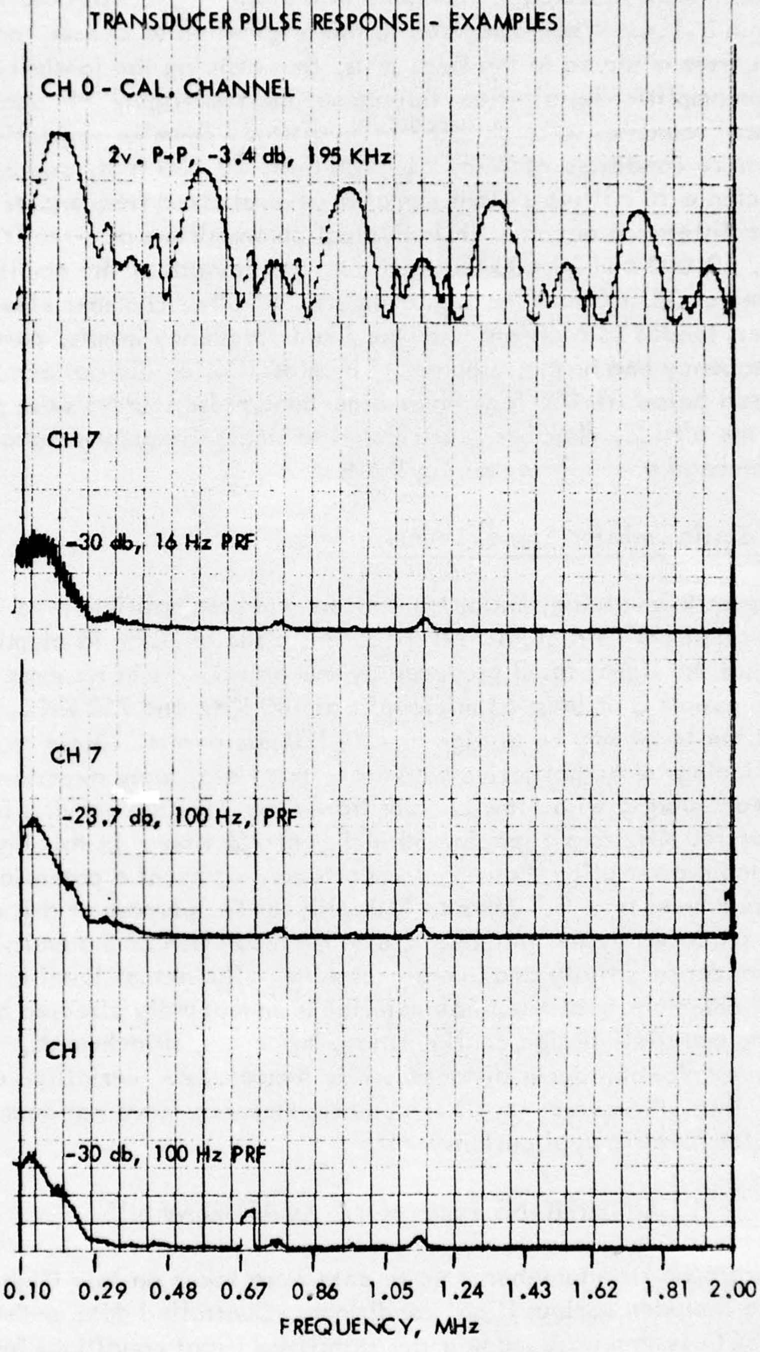


Figure 16. Examples of the AE/SCG Pre-Flight Calibration 140 KHz Pulse Response.

3.3 Post-Flight Calibration

After the last test flight, the AE/SCG system was checked by introducing a pulse near each installed transducer as done in the pre-flight tests described in Paragraph 3.1.4. When compared to the original pulse checks, no significant changes were observed in the final tests, thus assuring the integrity of the transducer/preamplifier installations throughout the test flights. In addition, the preamplifiers' responses were checked in an altitude chamber simulating temperature and pressure conditions at zero, 10,000 feet, 20,000 feet, and 35,000 feet by introducing a 10 millivolt input signal at several fixed frequencies. Very little response difference occurred in individual preamplifiers as a result of the conditions at zero, 10,000 and 20,000 feet altitude, but several of the amplifiers sporadically oscillated at 35,000 feet (no signal input). At other chamber elevations, the amplifiers tended to oscillate with the fixed-frequency inputs, particularly at the high frequency end of the response. Since oscillation did not occur during the flight tests below 31,000 feet when wide-band noise sources were present, oscillation in the altitude chamber is attributed to single-frequency signal injection into the wide-band preamplifier during the tests.

3.4 Acoustic Emission Signal Levels

Previous work involving similar transducers and preamplifiers used with the Flaw Locator on aircraft structural test specimens made of 7075-T6 aluminum alloy has shown that the signal level produced by the transducers in response to acoustic emission events is at least 65 microvolts at 140 KHz and 750 KHz. Indications are that the level may be as high as 100-120 microvolts. These levels, supported by the findings of authoritative workers in this field, were experienced at Lockheed-Georgia with Flaw Locator transducer spacings varying from 29 to 40 inches at 140 KHz and a spacing 30 inches at 750 KHz. In this light, it appears a reasonable assumption that 65 microvolts can represent a potential acoustic emission signal level from 0.1 MHz to 1.0 MHz for the purpose of this analysis. The voltage produced by the transducers as a manifestation of the emissions is affected by transducer sensitivity and overall response. The actual level produced by acoustic emissions in a structural material is undoubtedly affected by many factors, including material, design configuration, thickness, attachments, sound attenuation, source-to-transducer distance, wide temperature variations and possibly others. Thus, the actual levels of acoustic emissions have not been adequately defined for specific applications.

4.0 FLIGHT TEST CONDITIONS FOR NOISE MEASUREMENTS

The background structure-borne noise data were taken on four flights of C-5A LAC 0003 and included various flight conditions. Controlled data gathering runs of the AE/SCG system were made under stabilized flight conditions for a continuous duration of at least ten minutes each run. Data were produced during additional but noncontrolled runs. The AE system was allowed to operate continuously during each flight although strict attention was paid to the system only during the controlled runs. About 3 hours and 50 minutes of usable AE/SCG data were recorded during the four flights in controlled and noncontrolled tests. During these flights, the system was operated by a flight test engineer who was thoroughly familiar with operation of the equipment and its purposes. The SCG tests were incorporated into the flight plan prior to each flight to give them full cognizance. Flight notes were kept during flights to document conditions and events and to relate them to the recorder time code. Thus, a correlation can be made between the flight notes and the recorded SCG data. The conditions for the various flights are described in the following paragraphs.

4.1 The AE/SCG System

4.1.1 Flight 356, 9 July 1974

Purpose of the flight was to test the ALDCS, FSMS, and SCG systems. The AE/SCG system was operated for ten minutes during Flight Run 14 at a speed and altitude of 300 knots and 10,000 feet; Flight Run 15, ten minutes at stabilized speed and altitude of 300 knots and 35,000 feet; Flight Run 16, ten minutes at stabilized descent from 35,000 feet at 200 knots with engines 2 and 3 to reverse idle and engines 1 and 4 to idle. About 40 minutes of AE data were recorded during this flight.

4.1.2 Flight 357, 15 July 1974

The purpose of this flight was to test the ALDCS and FSMS systems under various conditions. The AE/SCG system was operated in uncontrolled tests and no continuous ten-minute runs were made. AE/SCG channels 1, 2,3 and 6 were not operated during the flight. Recorded AE/SCG data includes interrupted (often incomplete) transducer-channel scans at various flight conditions varying from 200-340 knots and from takeoff runs to 20,000 feet altitude, under cruise, slow sweeps, roller coasters and landing gear cycles. Close correlations between these flight events and individual transducer scans were not made. The preamplifier in Location Number 4 was found to be malfunctioning. About 20 minutes of useable data were recorded during this flight for Channels 5, 7, 8 and 9.

4.1.3 Flight 359, 17 July 1974

The purpose of this flight was to test the ALDCS, AE/SCG and other systems at various conditions. AE/SCG data were taken in controlled and uncontrolled tests. Continuous analyzer scans across all transducer channels were obtained on pre-takeoff and takeoff (Run 3) during 300 knot cruise and 10,000 feet (Run 5), during climb to and (300 knot) cruise at 20,000 feet, and during cruise maneuvers at 31-35,000 feet (Runs 12A1 thru 12B5). No SCG data were recorded during Run 14 which was a planned stabilized test run for AE/SCG at 10,000 feet and 300 knots. At least 1 hour 20 minutes of AE analyzer runs were made during this flight.

4.1.4 Flight 360, 19 July 1974

The purpose of this flight was to test the ALDCS and FSMS systems under various flight conditions and maneuvers. Dummy loads were placed on the inputs to transducer channels 7, 8, and 9 prior to flight to ground the input signals. The purpose for this was to determine the nature of extraneous signals, if any, produced in or induced into the preamplifier and cabling, which produced negative results. At 31,000 feet the remaining preamplifiers (1-6) appeared to be oscillating and were disconnected at the analyzer at that time. The conditions at which AE/SCG data were taken are: Run 5 - 1.2G to 2.1G coordinated turns at 220 knots and 10,000 feet; Run 10: 1.8G coordinated turns at 254 knots and 20,000 feet; Run 11: 1.8G coordinated turns at 262 knots and 29,000 feet; Run 12: 1.8G coordinated turns at 262 knots and 29,000 feet; and Run 13: 1.8G coordinated turns at 254 knots and 10,000 feet. At least 1.5 hours of AE analyzer runs were made during this flight. The structure-borne noise produced by the events of this flight had characteristics similar to the noise measured during controlled runs in other flights and no peculiarities in the data were noted. The results are therefore typified by the results of previous controlled runs.

> Note: Flight 358 was demonstration flight and neither the AE/SCG nor Flaw Locator systems were operated.

4.2 Flaw Locator System

The two Flaw Locator Systems were turned on at the beginning of each flight and slight adjustments were made to the gain settings after becoming airborne. The systems remained on continuously to the end of the flight. However, the contents of the Flaw Locator's memories were recorded on X-Y charts prior to cutting power at the end of flights. No time correlation was made for correlation to flight events; thus, the information stored by the Flaw Locators is integrated over the entire flight. The data accumulation time for the various flights are:

Flight 356	-	approximately 4 hours 45 minutes
Flight 357	-	approximately 3 hours 16 minutes
Flight 359	-	greater than 6 hours
Flight 360	-	approximately 4 hours

5.0 DATA REDUCTION

5.1 Data Sources

The data sources used in the analysis include:

- o Preflight SCG system calibration, recorded on magnetic tape
- o SCG system dynamic range, recorded on magnetic tape
- o Transducer response curves

- o Preamplifier response curves
- o SCG flight test data, recorded on magnetic tape for each flight
- o Test flight records

5.2 Ground Calibration and Flight Test Data

These data were recorded on magnetic tape and in each case included inputs from the AE spectrum analyzer signal channel, the VCO ramp voltage status, the frequency ramp, the channel identification status, and the time code, which were digitized in the PCM for recording. The tapes were later played through the ground-based data reduction facility to convert the digitized data to analog signals and then recorded on analog 8-track strip charts. The analog data were reduced to numerical voltage and decibels to show the signal levels which were actually produced at the analyzer and preamplifier inputs. The data from the preamplifiers were then corrected for transducer deviation from flatness to obtain voltage curves proportional to the true spectral noise level incident on the transducers.

5.2.1 Preflight Calibration Data Reduction

In the preflight calibration of the installed SCG system, the Channel 9 response was measured first from 0.1 to 2.0 MHz with both the short (< 10') and long (~ 300') preamplifier output cables to generate two response curves which differed by an amount depending on the frequency-dependent cable attenuation (Figure 14). The Channel 9 preamplifier gain was subtracted from the short-cable response to determine the spectral output of the random noise generator which was incident at the preamplifier input terminals. Using the resulting spectral noise input curve and the Channel 9 long-cable response, the system gain for the channel was determined as a function of frequency.

Since the calibration response curves for Channels 1 through 8, shown in Figure 15, were obtained for only the long-cable configuration, the spectral noise input curve as derived above was used as a baseline input signal in determining the system gain for each of these channels. The system gain data thus obtained for each channel are presented in Table C-1 of Appendix C. The system gain data were used in analysis of the SCG flight test data.

5.2.2 SCG Flight Test Data Reduction

The SCG flight test data consisted of spectral structure-borne noise detected at each of the nine locations during flight of the aircraft. Figure C-3 of Appendix C shows the strip chart recordings for Channels 2, 3, 4, and 9 at 10,000 and 20,000 feet. The noise inputs, recorded in terms of signal level at the analyzer, were converted to decibels with the use of Figure C-1 of Appendix C and are tabulated for Flights 356, 357, and 359 as a function of frequency in the tables in Appendix C. The channel system gain in decibels was subtracted from the noise signal to obtain the noise level at the preamplifier input for each channel.

Since the spectral input at this point is shaped in amplitude by the transducer response, a correction was made to compensate for the transducer deviation from a perfectly flat response. The corrections were made relative to the transducer response peak which was assigned a reference value of 0 decibels. All other points in the response then are considered losses relative to the peak and are thus assigned decibels of loss which are added back to the input noise signal. The values of these losses are tabulated in Table C-2 of Appendix C. The result is a spectral noise level which would be obtained if the transducer and SCG system responses had been perfectly flat over the frequency range of interest. In Figure 17 the three curves described above are shown for Channels 2, 3, 4, and 9 which were the Channels generally registering the highest levels of structure-borne noise at 10,000 and 20,000 feet.

Curve C in each chart is proportional to the true spectral structure-borne noise incident on the transducer. The proportionality is influenced by the absolute sensitivity of the transducer and the coupling factor of the transducer/structure interface. Since the nine transducers vary among themselves in absolute sensitivity, the constructed noise input curves are biased relative to one another due to the sensitivity differences. The sensitivity of each transducer is given on its response curve in Appendix A in terms of decibels below 1 volt per microbar of pressure (1v./μ bar). The bias can be eliminated by finding the relative sensitivity differences at each frequency for each of the transducers and using these values to compensate the constructed noise input curves. This was not done for three reasons: (1) the absence of a standard transducer, (2) the unknown relationship between the structure-borne noise and the transducer calibration source (spark-bar noise source), and (3) the unknown coupling factor for each transducer. However, the peak sensitivities of all transducers with the exception of the Channel 7 transducer are within 3.0 decibels of each other, so that the bias is generally not large, though it is significant. Most of the transducers' sensitivity peaks are at 150 or 200 KHz, except for deviations noted on Channel 5 and 7 transducers which peak near 400 KHz and near 500 KHz, respectively. A study of the transducer response curves in the Appendix will reveal these characteristics.

5.2.3 Flight Records

The handwritten records made during each flight were used to correlate the SCG taped data with aircraft altitude, speed, and maneuver. The common tie between the flight records and the magnetic tape was the time designation provided by the time code generator handwritten in numerals on the flight records and recorded in digital code on the tapes. The SCG flight data were taken primarily at stabilized flight conditions at altitudes of 10,000, 20,000, and 35,000 feet and at speeds not exceeding 340 knots at 10M and 20M feet and not exceeding 300 knots at 35M feet.

5.2.4 Flaw Locator Data Reduction

The Flaw Locator System was operated at a gain of approximately 86 db which

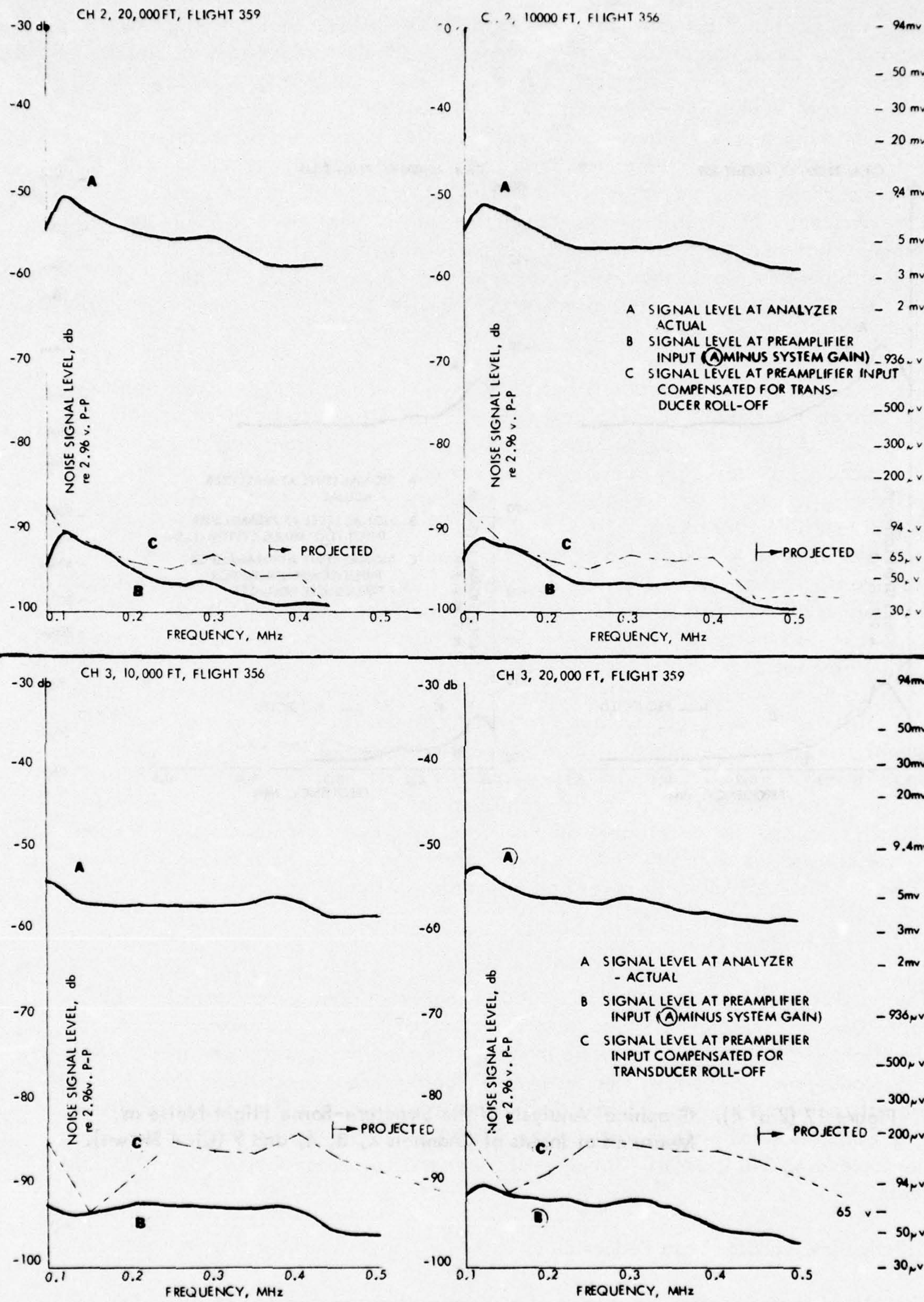


Figure 17 (1 of 4). Graphical Analysis of the Structure-Borne Flight Noise as Measured at Inputs of Channels 2, 3, 4 and 9 (CH 2 and 3 Shown).

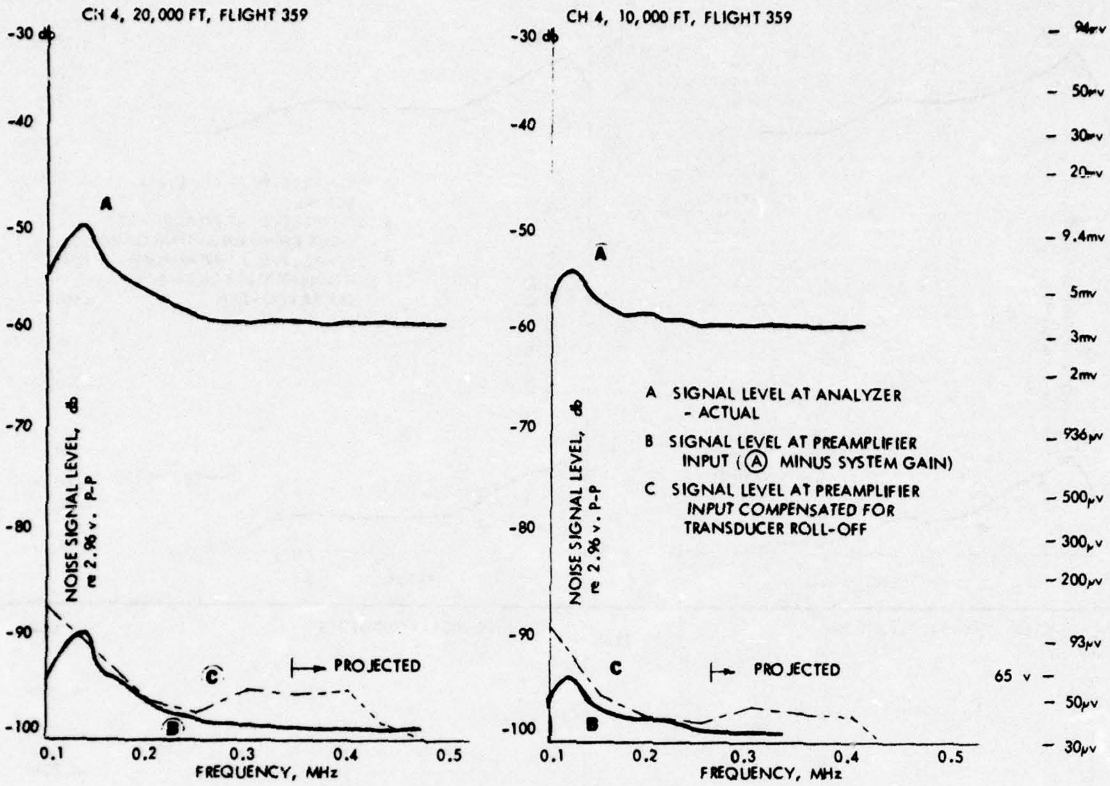
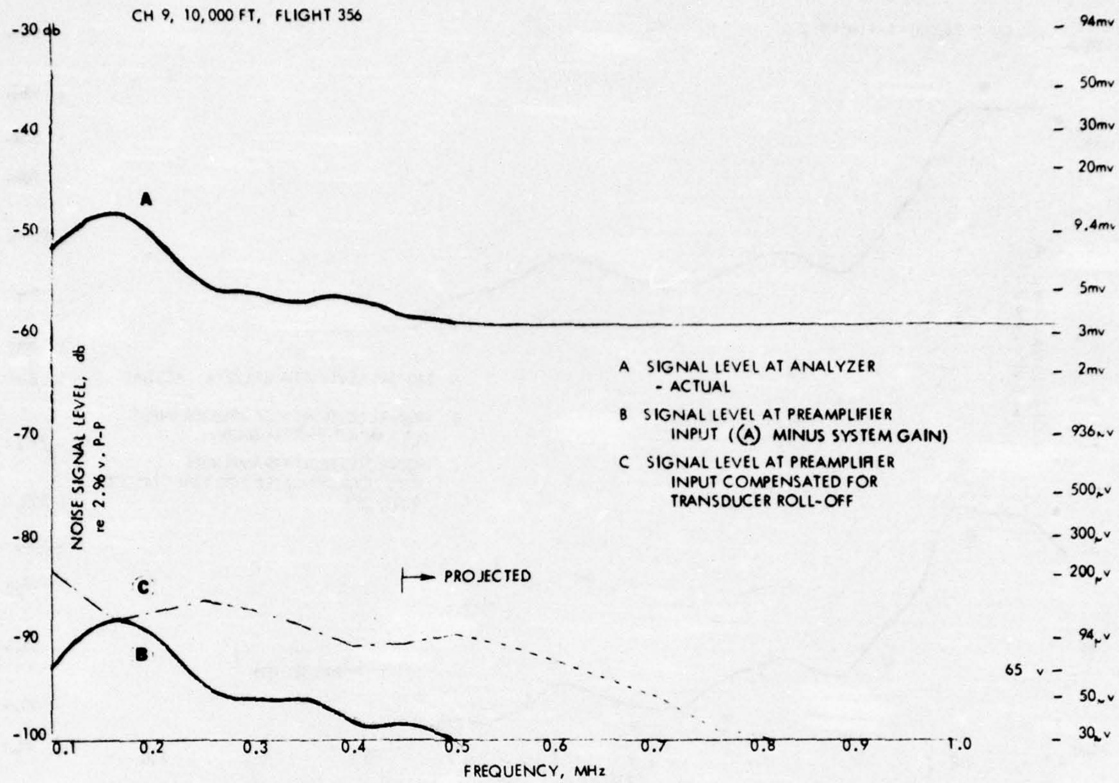


Figure 17 (2 of 4). Graphical Analysis of the Structure-Borne Flight Noise as Measured at Inputs of Channels 2, 3, 4, and 9 (Ch 4 Shown).



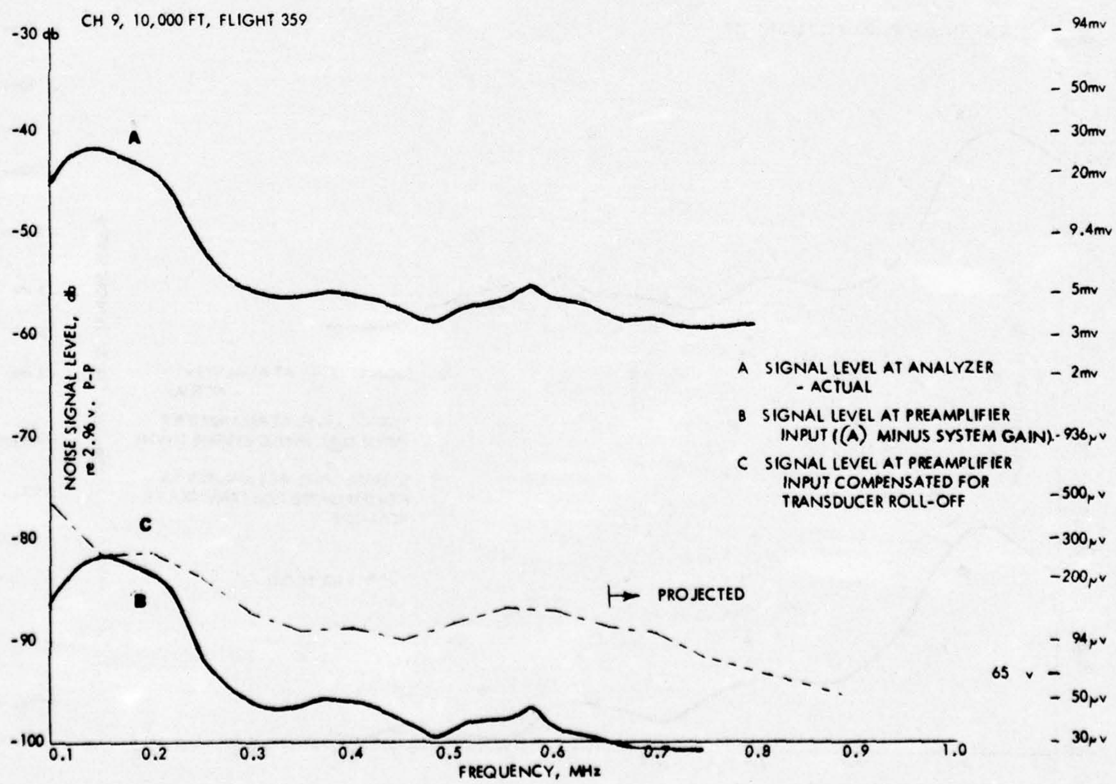
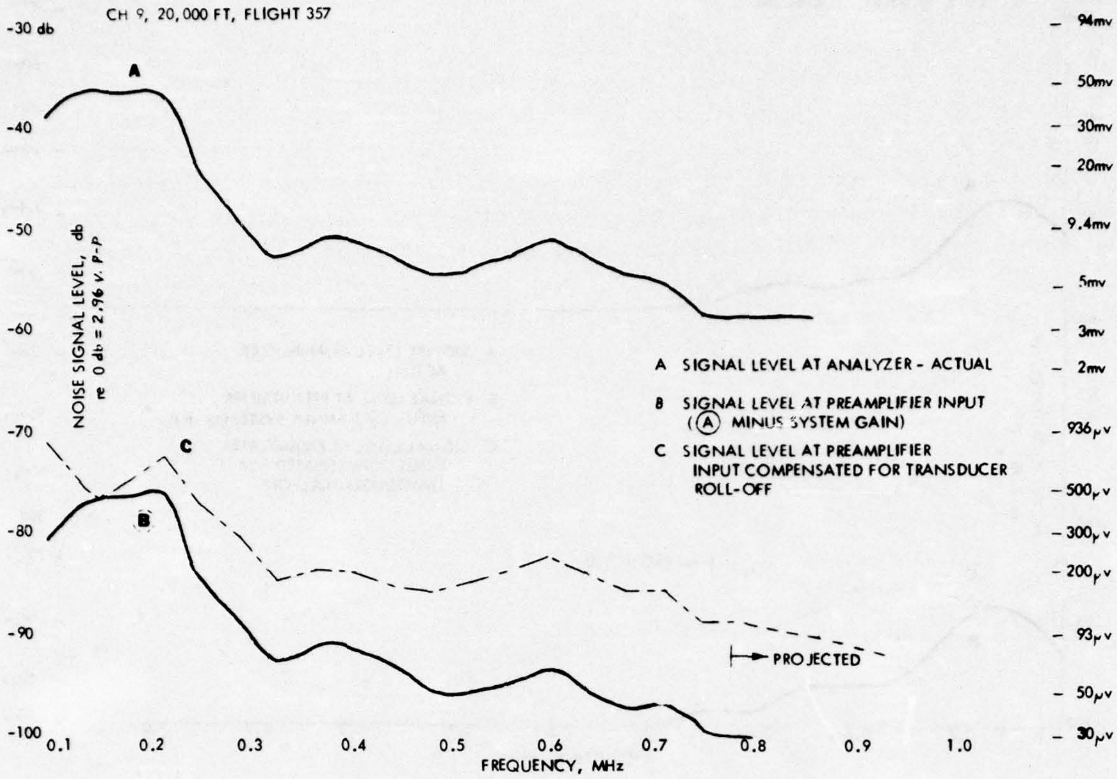


Figure 17 (3 of 4). Graphical Analysis of the Structure-Borne Flight Noise as Measured at Inputs of Channels 2, 3, 4 & 9 (Ch 9 Shown)



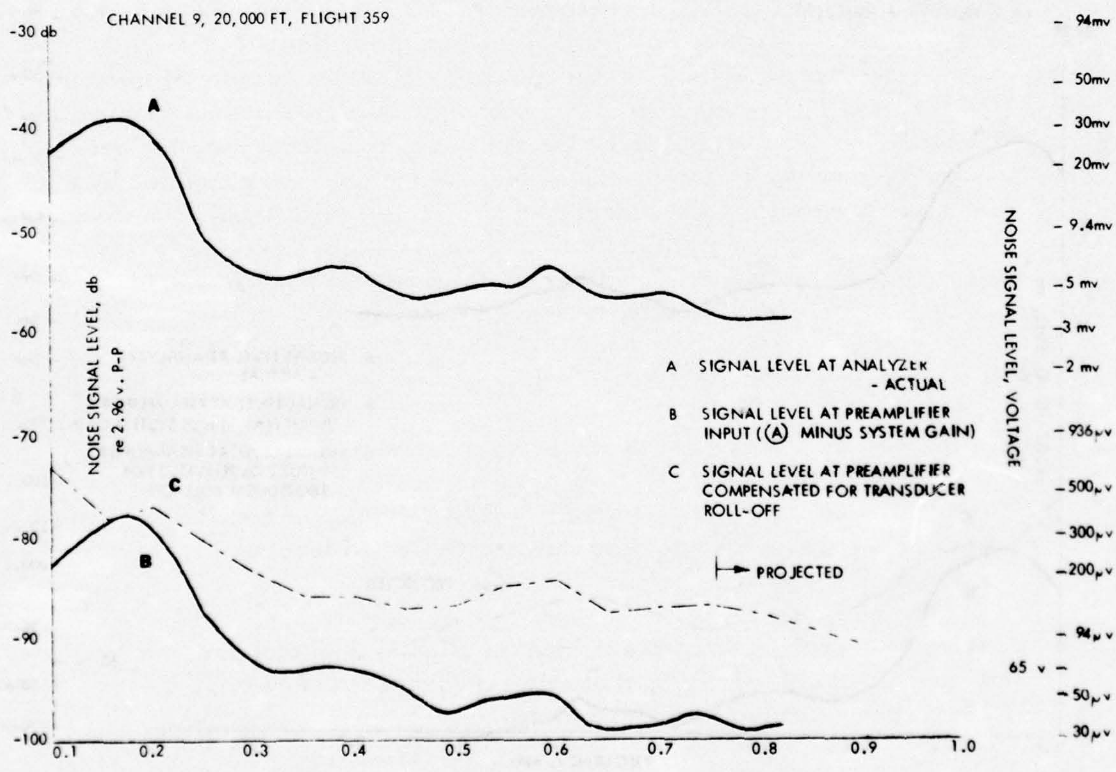


Figure 17 (4 of 4). Graphical Analysis of the Structure-Borne Flight Noise as Measured at Inputs to Channels 2, 3, 4 and 9 (Ch 9 Shown).

provided a system sensitivity of 50 microvolts. Noise signals below this amplitude would not be able to trigger the Flaw Locator's counting circuits. The system was operated for four or more hours on each of four flights. Both Flaw Locator units in the system registered noise counts which were distributed sparsely and randomly indicating that the structure-borne noise in the attachment area was very low and would not be sufficient to hamper slow crack growth detection. The results from one of the flights is shown in Figure 18 at two recording sensitivities. The noise is so low that individual noise counts are easily distinguishable. The Flaw Locator data was also looked at for evidence of crack growth during flight and verification by standard NDT methods. No indications were present that could be considered significant in this respect. No indications were of sufficient amplitude above the noise, considering the duration of the monitoring, and there was no consistent build-up of counts from flight-to-flight in a specific memory location to indicate the presence of an active crack.

6.0 RESULTS AND CONCLUSIONS

6.1 Results

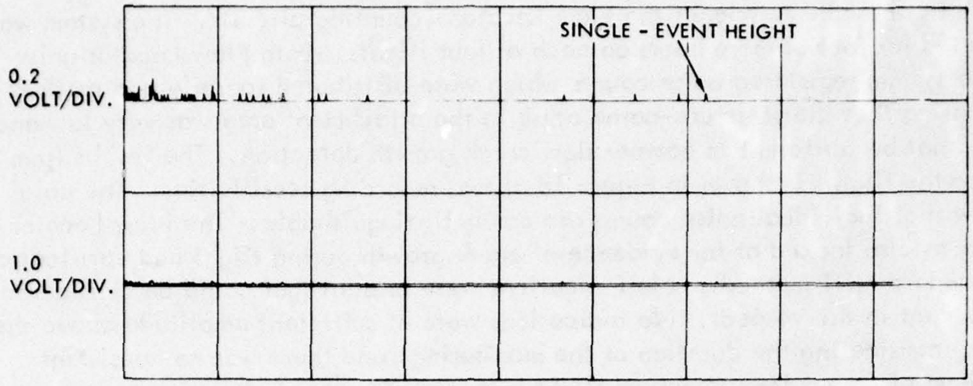
Results of the AE/SCG structure-borne flight noise survey are illustrated graphically in Figure 19 and summarized in Table I. The Figure shows the noise envelope curve for the channels generally registering the highest levels of noise. They were constructed from the flight test and calibration data. The curves have been corrected for transducer and system response nonlinearity and are proportional to the noise incident on the transducer. A "bias" exists among the channels because of transducer sensitivity differences. The table lists the highest noise measured in each channel at 10,000 and 20,000 feet altitudes and a frequency below which a hypothetical signal-to-noise (an assumed acoustic emission level which could be generated by a slowly growing crack versus the structure-borne noise level) ratio equals 2:1 at some point. The hypothetical signal-to-noise ratio has been discussed previously. No data is presented for the system at 35,000 feet because the preamplifiers were oscillating at this altitude and measurements could not be made.

The test results can be summarized by the following statements:

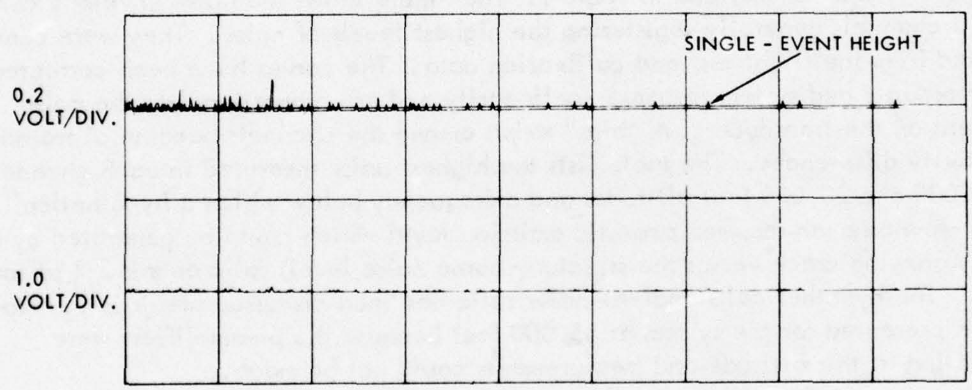
1. Structure-borne noise varies with location on the aircraft.

2. Measured noise levels are relative to overall response of the system and particularly to transducer sensitivity and spectral response.

3. Structure-borne noise appears to increase slightly with altitude at some structural reactions, probably because of structure and/or system changes at reduced temperatures.



LOCATOR NO. 1



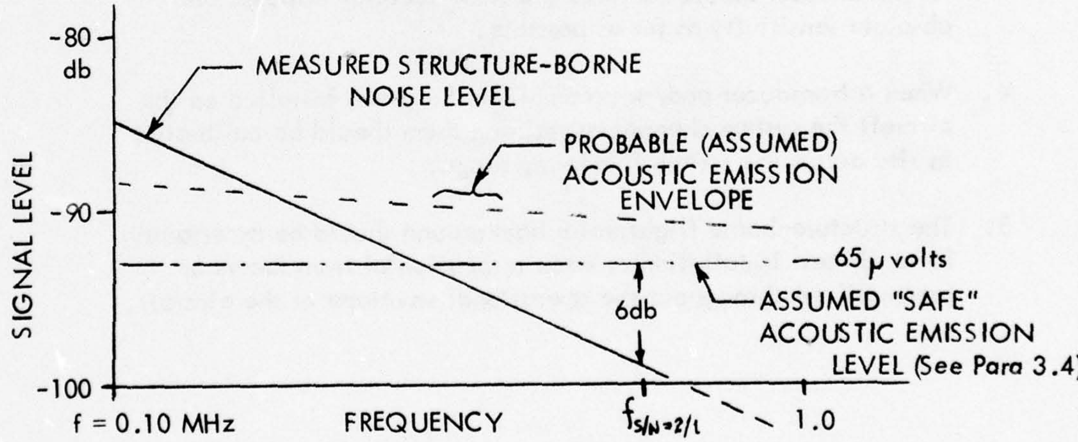
LOCATOR NO. 2

Figure 18. Graphical Results of the Flaw Locator Structure-Borne Flight Noise Monitoring on the Center Wing. X-Axis Represents Position Along Straight Line Between Two Transducers.

TABLE I
SUMMARY MATRIX OF STRUCTURE-BORNE NOISE MEASUREMENTS

CHANNEL	NOISE LEVELS - 10,000 FT.			NOISE LEVELS, 20,000 FT.		
	MAX NOISE	FREQ	FREQ WHERE S/N = 2/1	MAX NOISE	FREQ	FREQ WHERE S/N = 2/1
1	300 μvolts	0.10 MHz	≥ 0.50 MHz	300 μvolts	0.10 MHz	≥ 0.45 MHz
2	130	0.10	≥ 0.50	120	0.10	≥ 0.50
3	170	0.10	≥ 0.70	190	0.10	≥ 0.70
4	120	0.10	≥ 0.45	330	0.10-0.14	≥ 0.45
5	190	0.10	≥ 0.35	190	0.10	≥ 0.35
6	120	0.10	≥ 0.35	100	0.10	≥ 0.40
7*		0.10	≥ 0.40		0.10	≥ 0.40
8	180	0.10	≥ 0.45	180	0.10-0.14	≥ 0.45
9	380	0.10	≥ 0.95	860	0.10	≥ 1.0

*Insufficient Transducer Response Data at Lower Frequencies



$f_{S/N=2/1}$ = frequency where signal to noise ratio equals 2:1

$$6 \text{ db} = 20 \log_{10} \text{Signal/Noise} = 20 \log_{10} 2.0$$

Figure 19. Summary Tabulation of the AE/SCG Structure-Borne Flight Noise Measurement Results.

4. Below 400 KHz, structure-borne noise is generally too high to achieve an acceptable signal-to-noise ratio for an acoustic emission SCG system.

5. The signal-to-noise ratio appears to increase to a value of 2:1 within the frequency range 400 to 750 KHz for all locations except the landing gear area (Channel 9) where the frequency ranges up to 1000 KHz.

6.2 Conclusions

The following statements represent conclusions based on the tests relative to future acoustic emission SCG systems installed on aircraft. The reference aircraft is a C-5A Galaxy transport.

1. Structure-borne noise levels on aircraft are not too high for operation of a practical AE/SCG system or commercial Flaw Locator system during flight, except below 400 KHz.

2. A practical AE/SCG system for the aircraft should be designed to operate at no lower frequency than 500 KHz for most structural locations. High-noise areas may require operation at 1.0 MHz to achieve an acceptable signal-to-noise ratio. An optimum operating frequency appears to be 750 KHz.

3. Transducers selected for a given AE/SCG system for installation on the aircraft should all have the same spectral response and absolute sensitivity as far as possible.

4. When a transducer and/or preamplifier has been installed on the aircraft the system channel containing them should be calibrated in situ across the system frequency range.

5. The structure-borne flight noise background should be ascertained for each new installation or each relocation of transducers or preamplifiers throughout the operational envelope of the aircraft.

REFERENCES

1. C. D. Bailey and W. M. Pless, "Acoustic Emission Used to Nondestructively Determine Crack Locations in Aircraft Structural Fatigue Specimen", Proceedings of Ninth Symposium on Nondestructive Evaluation, San Antonio, Texas, 25 April 1973.

2. C. D. Bailey and W. H. Sproat, "C-5A Flight Structural Monitoring System (FSMS), Final Report, Phase I-Feasibility", Appendix, p. A-1, Lockheed-Georgia Report LG73ER-0171, 27 September 1973, Contract AF 33(657)-15053 (SA 1000).

3. C. D. Bailey and W. H. Sproat, "C-5A Determination of Detectability Limits, Task H of Phase II, Flight Structural Monitoring System", Lockheed-Georgia Report LG74-0044, Contract AF 33(657)-15053 (Change Order P00832), 26 March 1974.

APPENDIX A - CALIBRATION DATA

This Appendix contains the calibration or response curves for the transducers and preamplifiers used in the AE/SCG system and the Flaw Locator transducers. Table I is also included to provide the serial numbers of the units installed in specific locations on the aircraft.

The identification and location of the preamplifier/transducer sets as installed on LAC-0003 are given in the following Table.

TABLE A1

IDENTIFICATION AND LOCATION OF PREAMPLIFIERS AND TRANSDUCERS ON LAC-0003

Location	Preamplifier Model 801P* Serial No.	Transducer Model D9201* Serial No.
1	80058	8A06
2	80309	8A34
3	80314	8A24
4	80283**	8A03
5	80057	8A46
6	80300	8A28
7	80281	8A36
8	80059	8A04
9	80302	8A40

*Dunegan/Endevco

**Replacement preamplifier after Flight #357

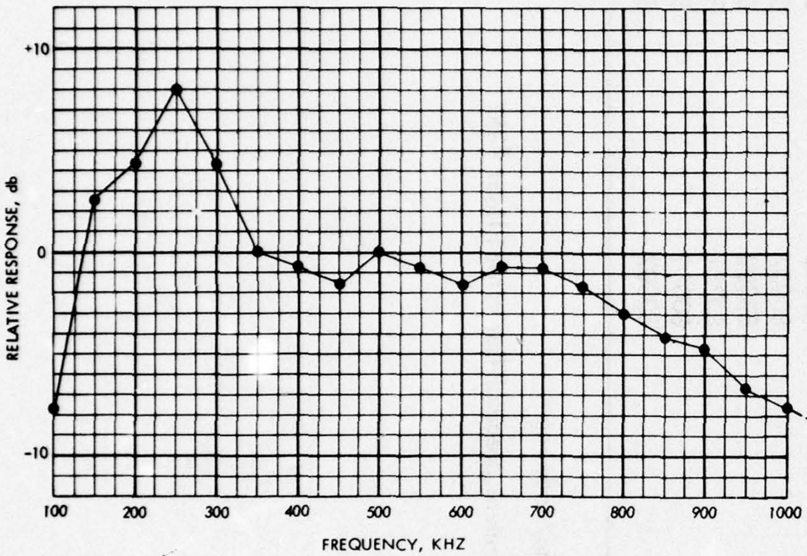
DUNEGAN ENDEVCO

Calibration Certificate

TRANSDUCER MODEL D9201 SERIAL NO. HAC6

SENSITIVITY (AVE.) -71 db re 1V/μbar

CAPACITY 136 pF



DATA WAS OBTAINED BY THE SPARK BAR CALIBRATION METHOD AS DESCRIBED BY DE 73-3

P/N 19022

DUNEGAN ENDEVCO

BY M.U.

DATE 1/7/74

DUNEGAN ENDEVCO

Calibration Certificate

TRANSDUCER MODEL D9201 SERIAL NO. HAC6

SENSITIVITY ______ db re 1V/μbar

CAPACITY ______ pF

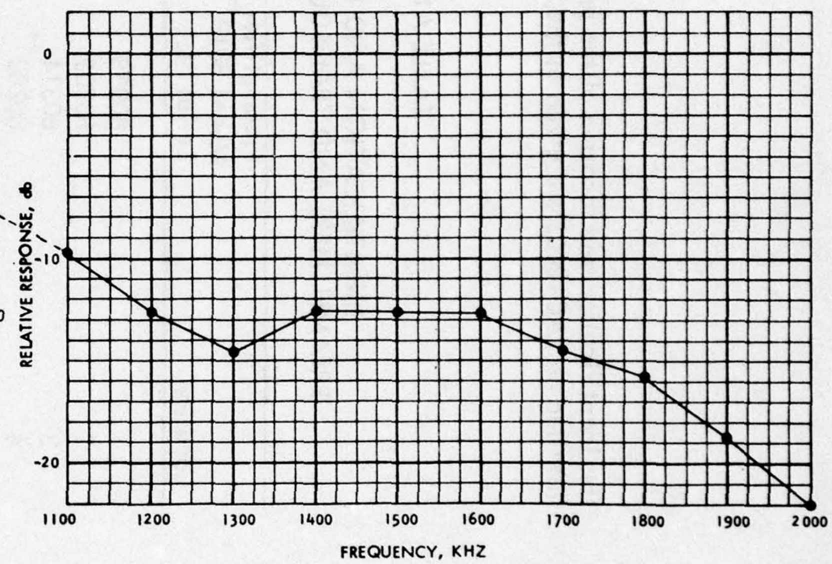


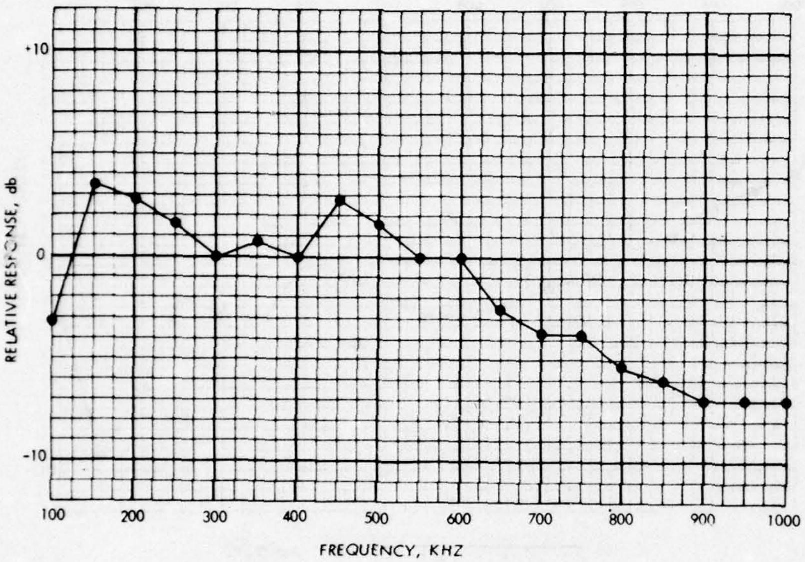
Figure A1 (1 of 9). AE/SCG Transducer Response. Ch 1.

Calibration Certificate

TRANSDUCER MODEL D9201 SERIAL NO. AA34

AVERAGE ~~MAXIMUM~~ SENSITIVITY -83 db re 1V/μbar

CAPACITY 482 482 247



DATA WAS OBTAINED BY THE SPARK BAR CALIBRATION METHOD AS DESCRIBED BY DE 73-3

BY

DATE

P/N 19022

Calibration Certificate

TRANSDUCER MODEL D9201 SERIAL NO. AA34

SENSITIVITY ______ db re 1V/μbar

CAPACITY ______ pF

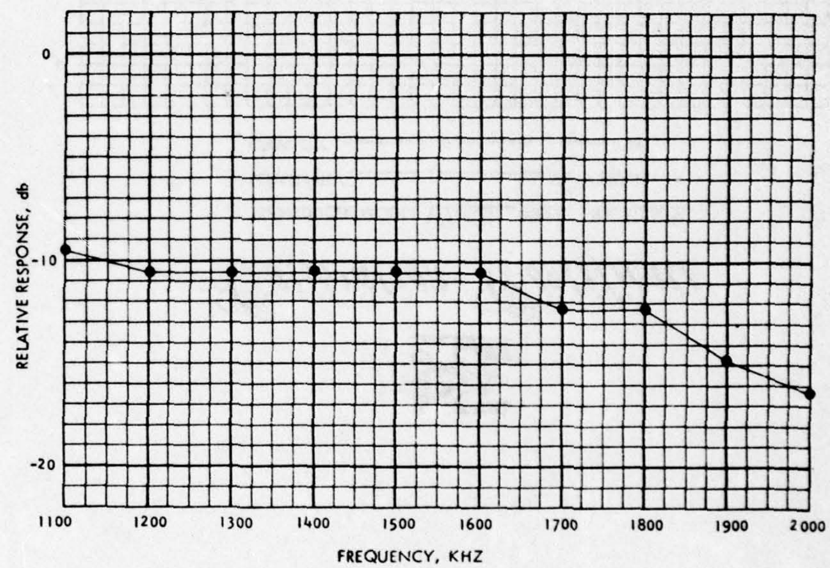


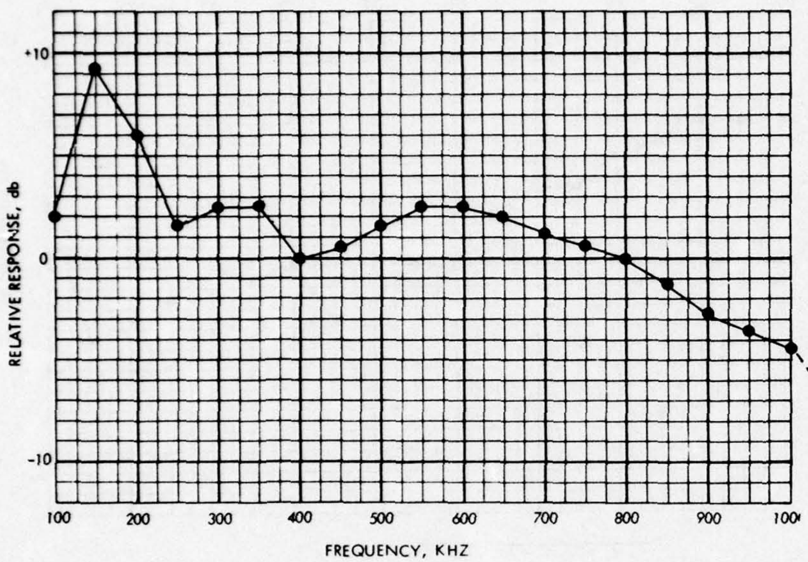
Figure A1 (2 of 9). AE/SCG Transducer Response. Ch 2.

Calibration Certificate

TRANSDUCER MODEL D9201 SERIAL NO. AA24

SENSITIVITY ________ -58 db re 1V/ubar

CAPACITY ________ pF



DATA WAS OBTAINED BY THE SPARK BAR CALIBRATION METHOD AS DESCRIBED BY DE 73-3

BY ________

DATE ________

P/N 19022

Calibration Certificate

TRANSDUCER MODEL D9201 SERIAL NO. AA24

SENSITIVITY ________ db re 1V/ubar

CAPACITY ________ pF

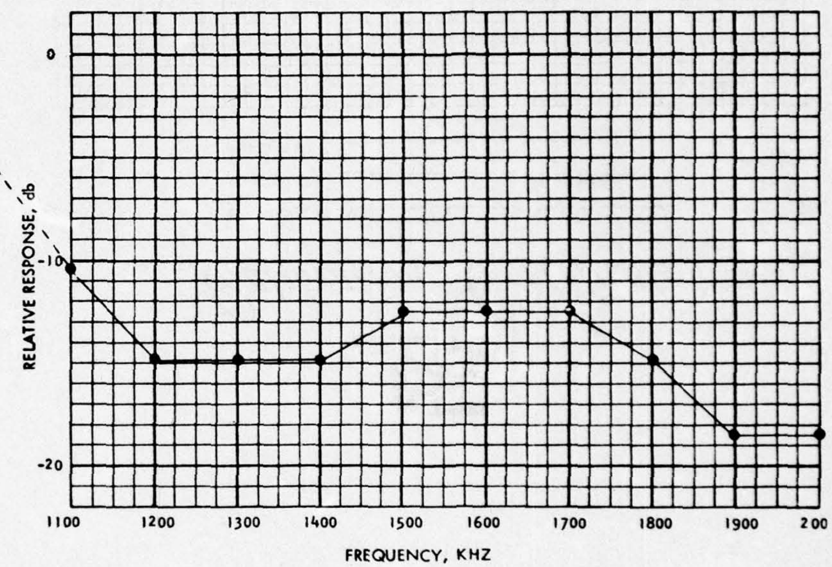


Figure A1 (3 of 9). AE/SCG Transducer Response. Ch 3.

DUNEGAN ENDEVCO

Calibration Certificate

TRANSDUCER MODEL D9201 SERIAL NO. AA03

SENSITIVITY -84 db re 1V/ubar

CAPACITY 452-461-246

DATA WAS OBTAINED BY THE SPARK BAR CALIBRATION METHOD AS DESCRIBED BY DE 73-3

P/N 19022

DUNEGAN ENDEVCO

BY ______

DATE 12-17-73

DUNEGAN ENDEVCO

Calibration Certificate

TRANSDUCER MODEL D9201 SERIAL NO. AA03

SENSITIVITY ______ db re 1V/ubar

CAPACITY ______ pF

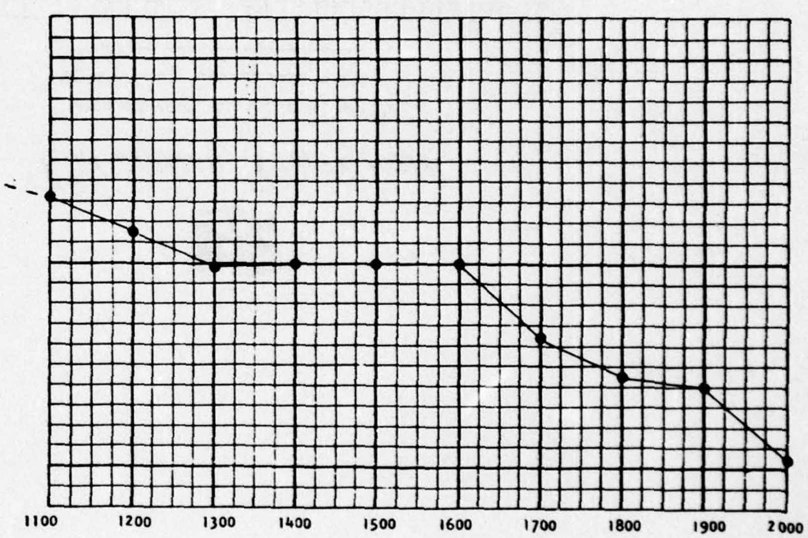


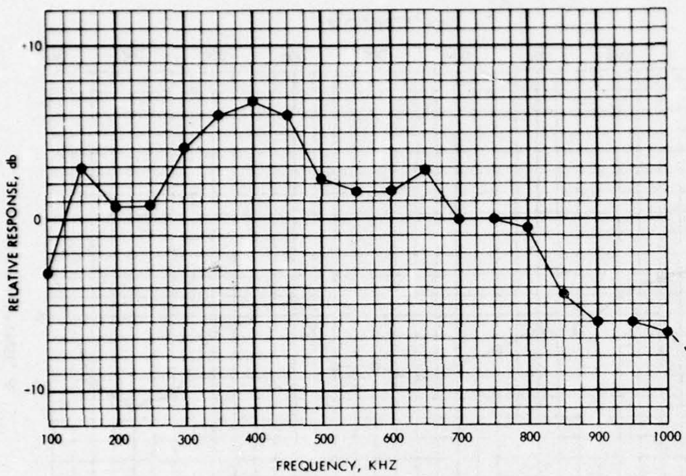
Figure A1 (4 of 9). AE/SCG Transducer Response. Ch 4.

Calibration Certificate

TRANSDUCER MODEL D9201 SERIAL NO. AA46

SENSITIVITY (AE-6) -56 db re 1V/μbar

CAPACITY _____ pF



DATA WAS OBTAINED BY THE SPARK BAR CALIBRATION METHOD AS DESCRIBED BY DE 73-3

BY _____

DATE 1/23/79

P/N 19022

Calibration Certificate

TRANSDUCER MODEL D9201 SERIAL NO. AA46

SENSITIVITY _____ db re 1V/μbar

CAPACITY _____ pF

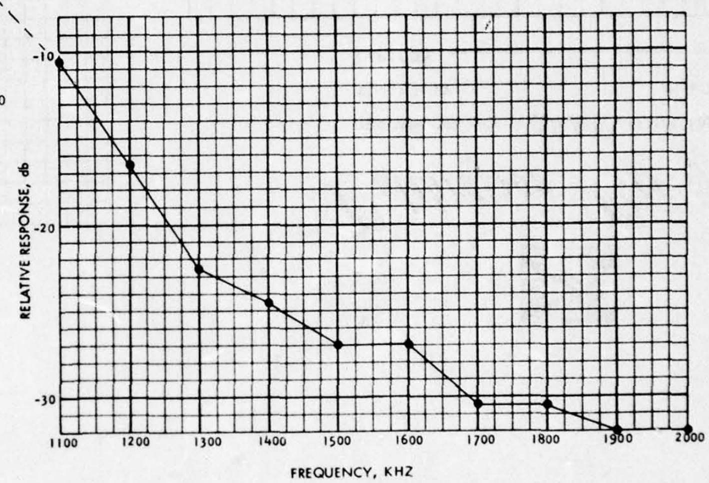


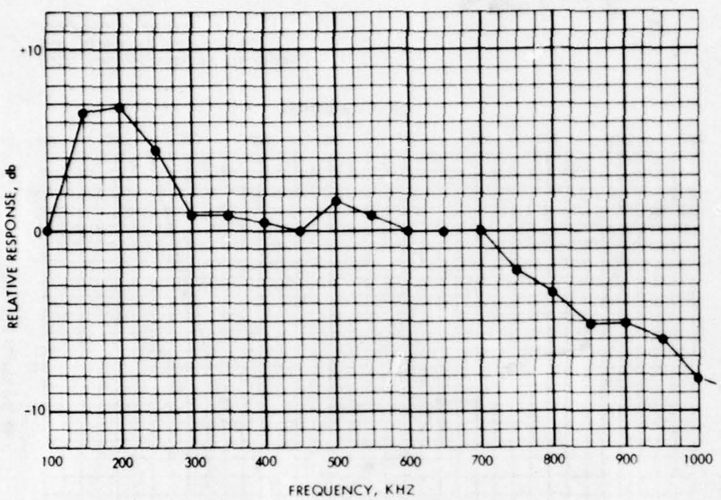
Figure A1 (5 of 9). AE/SCG Transducer Response. Ch 5.

Calibration Certificate

TRANSDUCER MODEL D9201 SERIAL NO. AA25

~~MAXIMUM~~ AVERAGE SENSITIVITY -84 db re 1V/µbar

CAPACITY 488-486-245



DATA WAS OBTAINED BY THE SPARK BAR CALIBRATION METHOD AS DESCRIBED BY DE 73-3

BY

DATE

P/N 19022

Calibration Certificate

TRANSDUCER MODEL D9201 SERIAL NO. AA25

SENSITIVITY ______ db re 1V/µbar

CAPACITY ______ pF

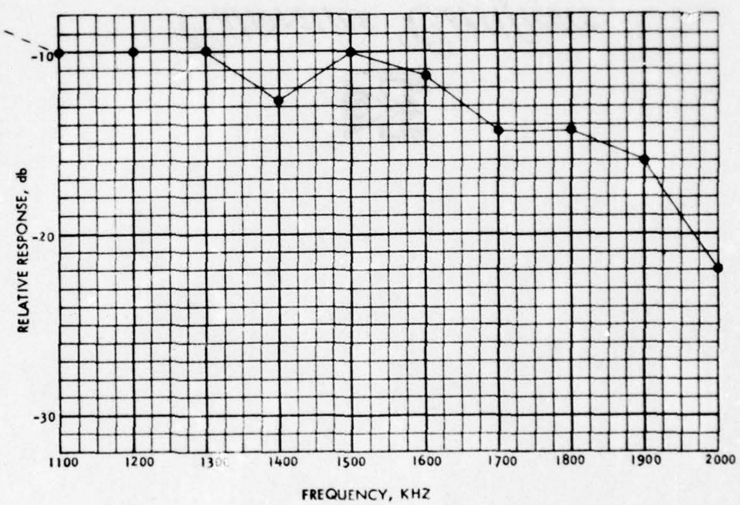


Figure A1 (6 of 9). AE/SCG Transducer Response. Ch 6.

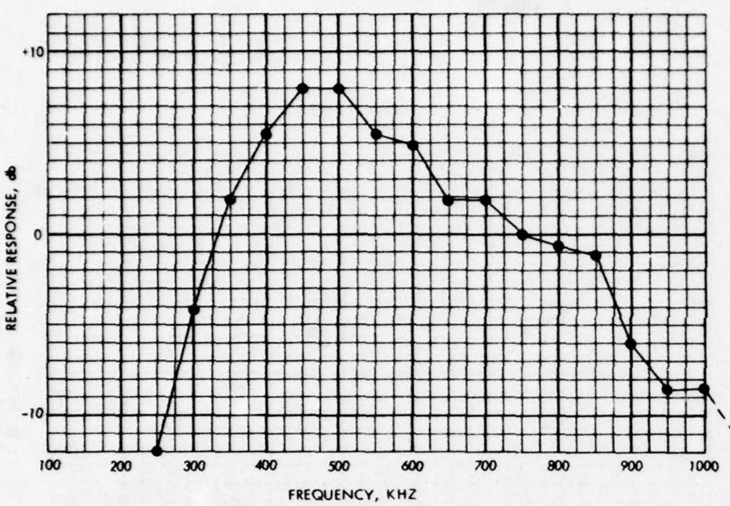
DUNEGAN ENDEVCO

Calibration Certificate

TRANSDUCER MODEL [illegible] SERIAL NO. [illegible]

SENSITIVITY ([illegible]) -72 db re 1V/μbar

CAPACITY ________ pF



DATA WAS OBTAINED BY THE SPARK BAR CALIBRATION METHOD AS DESCRIBED BY DE 73-3

DUNEGAN ENDEVCO

BY [illegible]

DATE 1/23/74

P/N 19022

DUNEGAN ENDEVCO

Calibration Certificate

TRANSDUCER MODEL D9201 SERIAL NO. [illegible]

SENSITIVITY ________ db re 1V/μbar

CAPACITY ________ pF

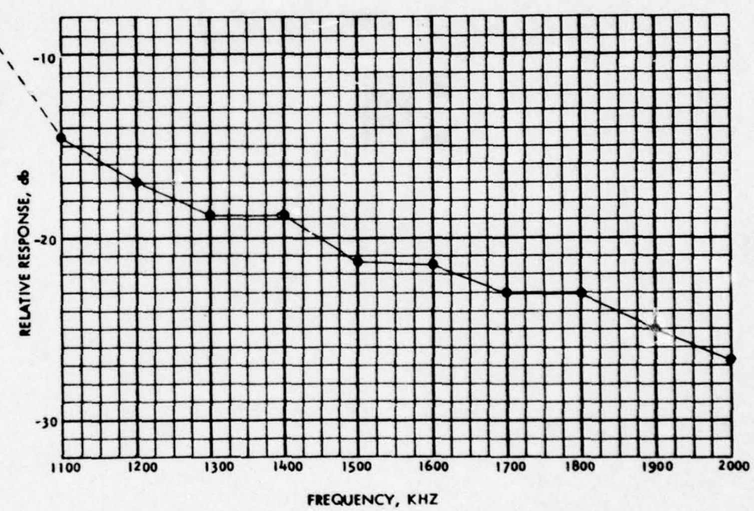


Figure A1 (7 of 9). AE/SCG Transducer Response. Ch 7.

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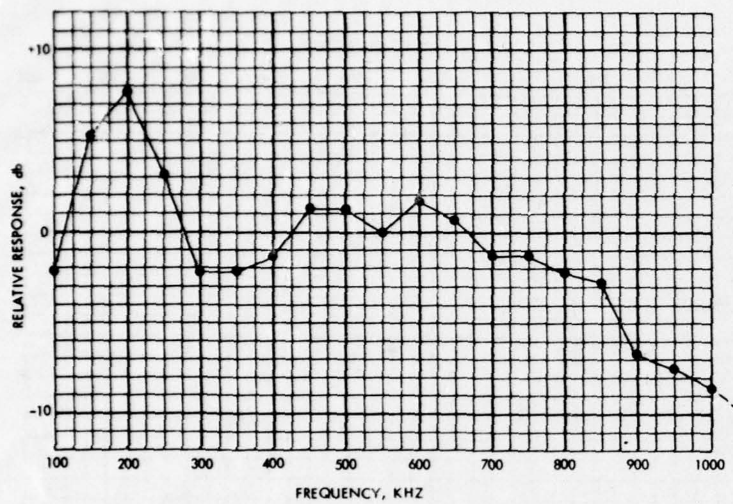
Calibration Certificate

TRANSDUCER MODEL D9201 SERIAL NO. AA04
[redacted] SENSITIVITY -85 db re 1V/μbar
CAPACITY 465-465-237



DATA WAS OBTAINED BY THE SPARK BAR CALIBRATION METHOD AS DESCRIBED BY DE 73-3

DUNEGAN ENDEVCO
BY
DATE 12-17-73

P/N 19022

DUNEGAN ENDEVCO

Calibration Certificate

TRANSDUCER MODEL D9201 SERIAL NO. AA04
SENSITIVITY ______ db re 1V/μbar
CAPACITY ______ pF

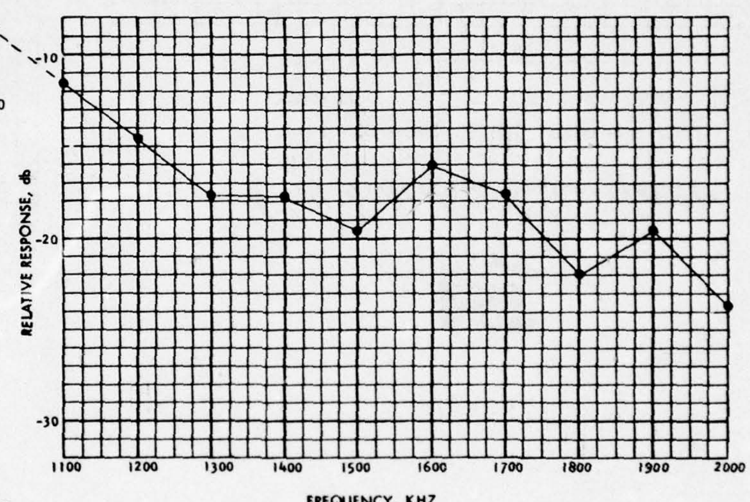


Figure A1 (8 of 9). AE/SCG Transducer Response. Ch 8.

DUNEGAN ENDEVCO

Calibration Certificate

TRANSDUCER MODEL D9201 SERIAL NO. A440

SENSITIVITY (AVG.) [illegible] db re 1V/μbar

CAPACITY [illegible] pF

DATA WAS OBTAINED BY THE SPARK BAR CALIBRATION METHOD AS DESCRIBED BY DE 73-3

DUNEGAN ENDEVCO

BY [illegible]

DATE 1/17/74

P/N 19022

DUNEGAN ENDEVCO

Calibration Certificate

TRANSDUCER MODEL D9201 SERIAL NO. A440

SENSITIVITY ______ db re 1V/μbar

CAPACITY ______ pF

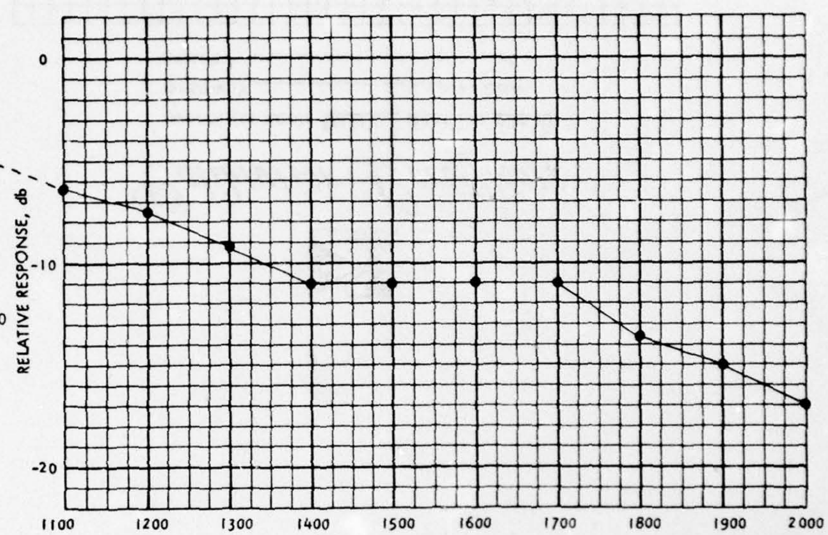


Figure A1 (9 of 9). AE/SCG Transducer Response. Ch 9.

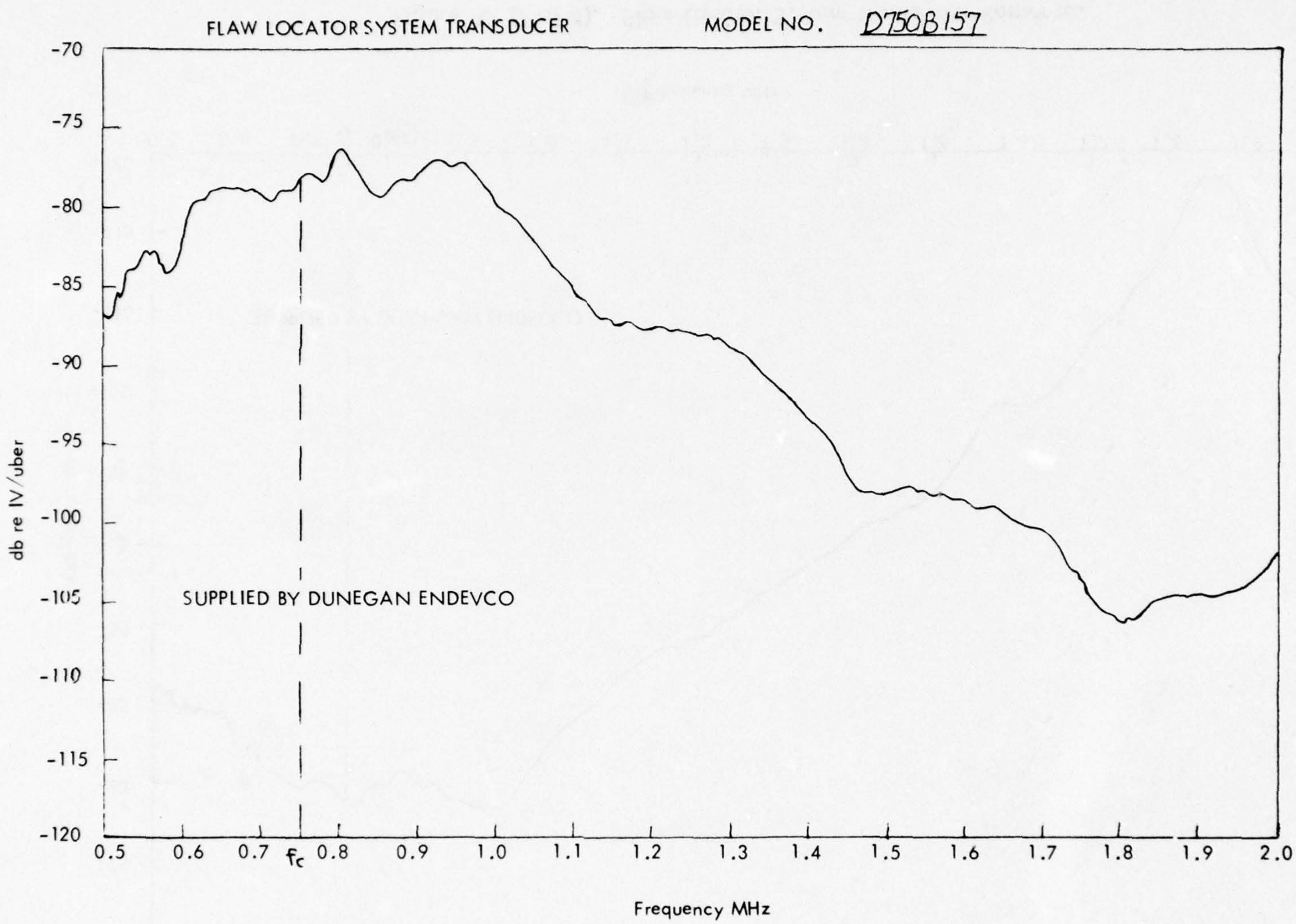


Figure A2 (1 of 4). Flaw Locator System Transducer Response

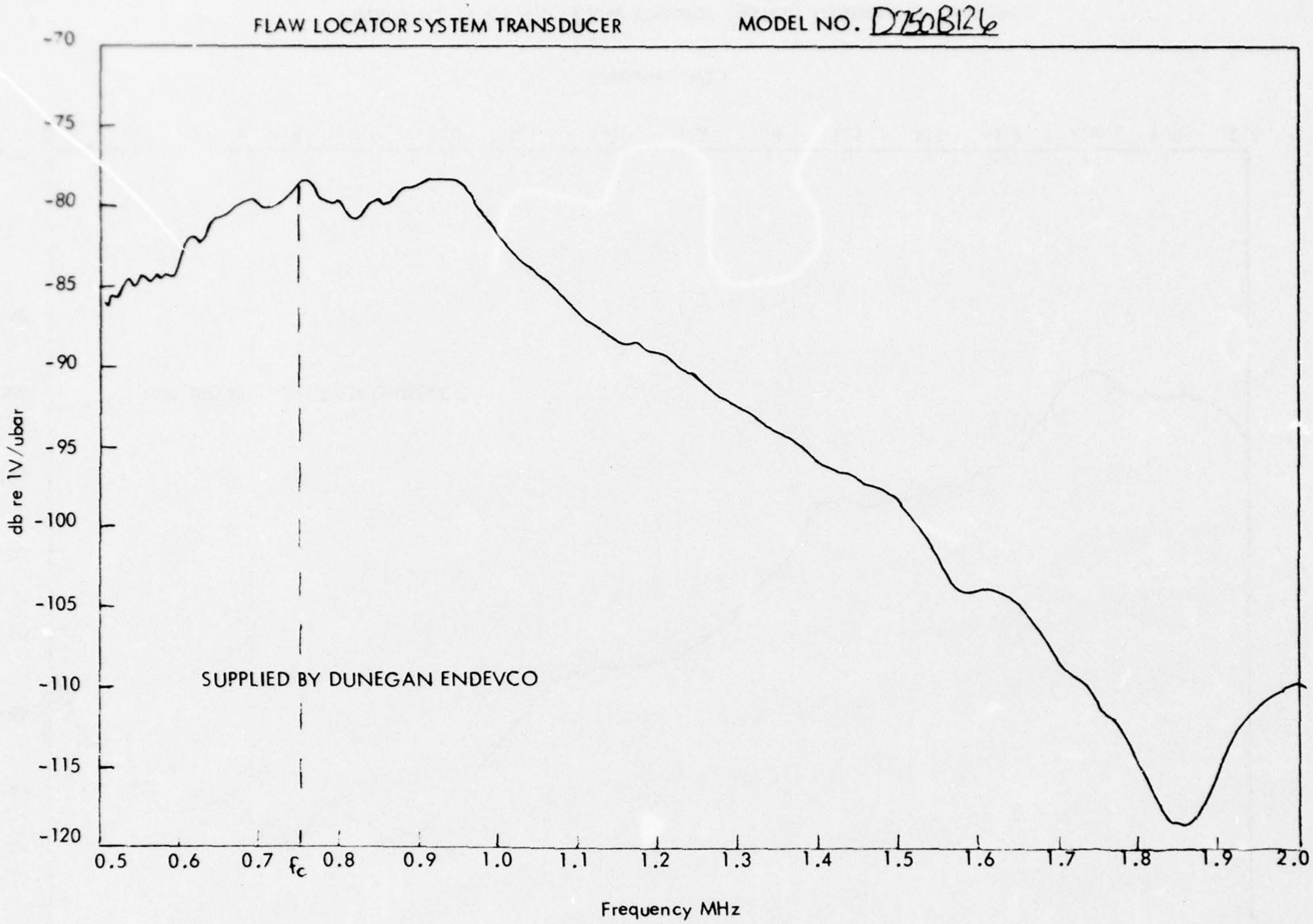


Figure A2 (2 of 4). Flaw Locator System Transducer Response.

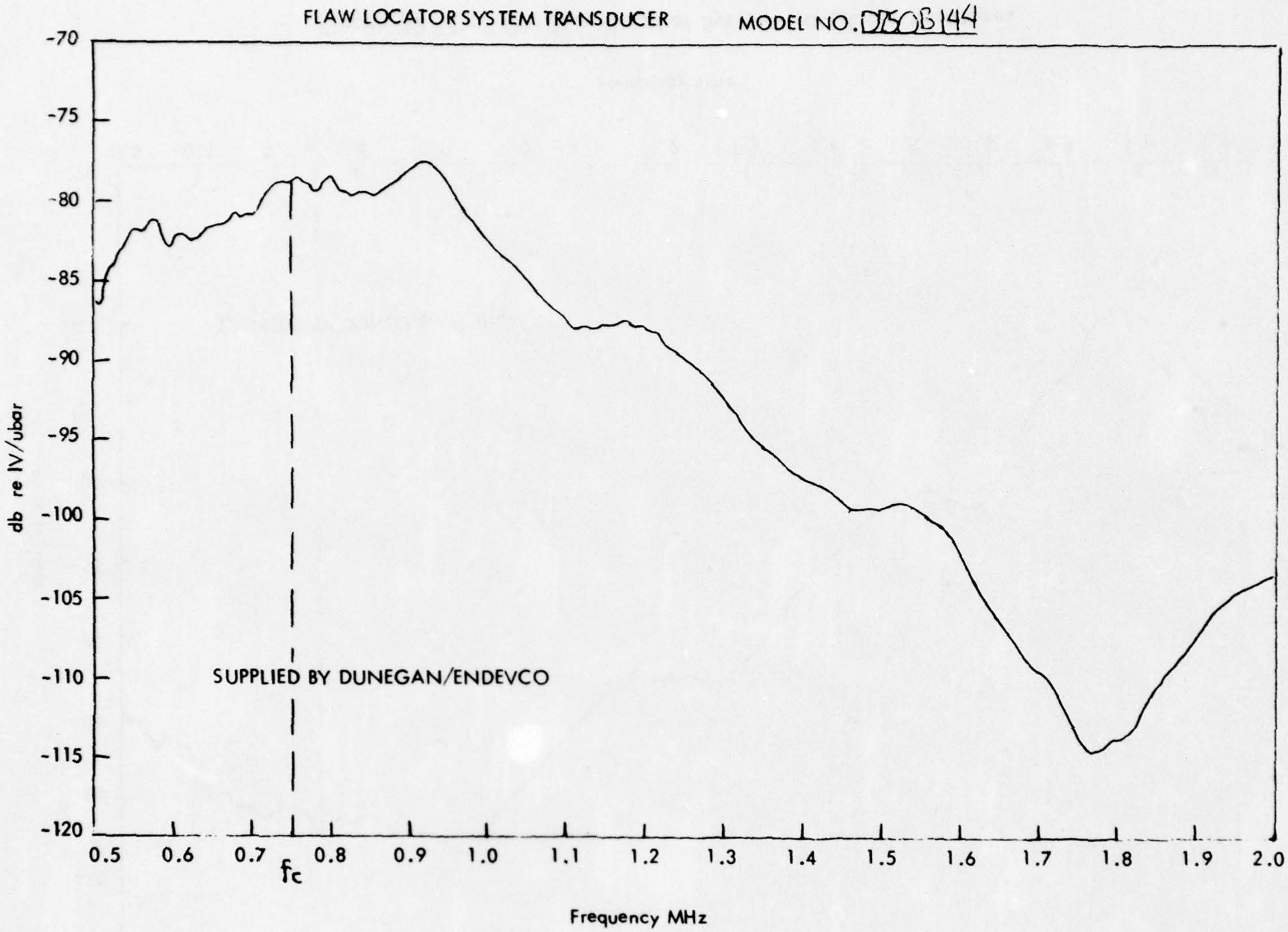


Figure A2 (3 of 4). Flaw Locator System Transducer Response.

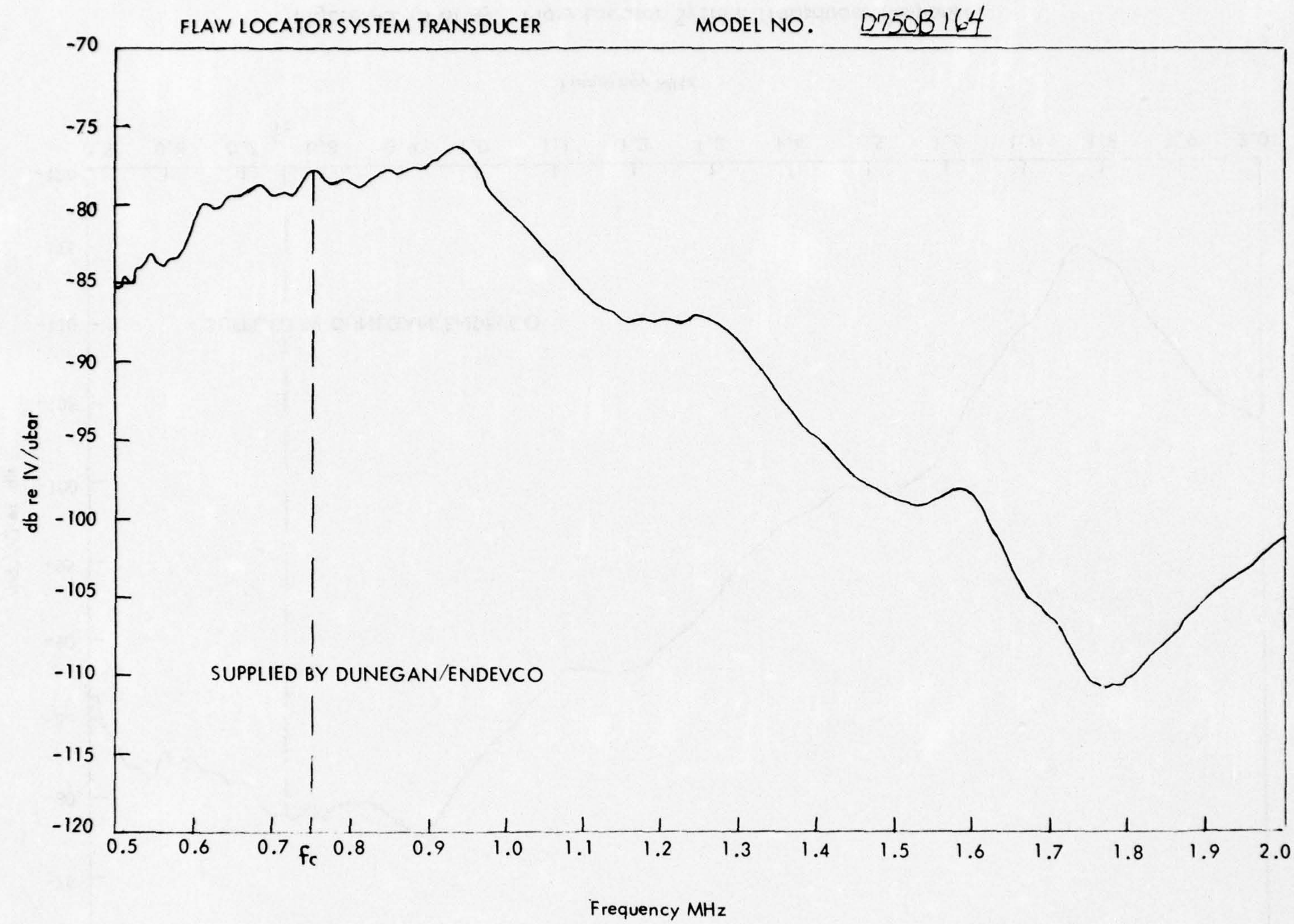


Figure A2 (4 of 4). Flaw Locator System Transducer Response.

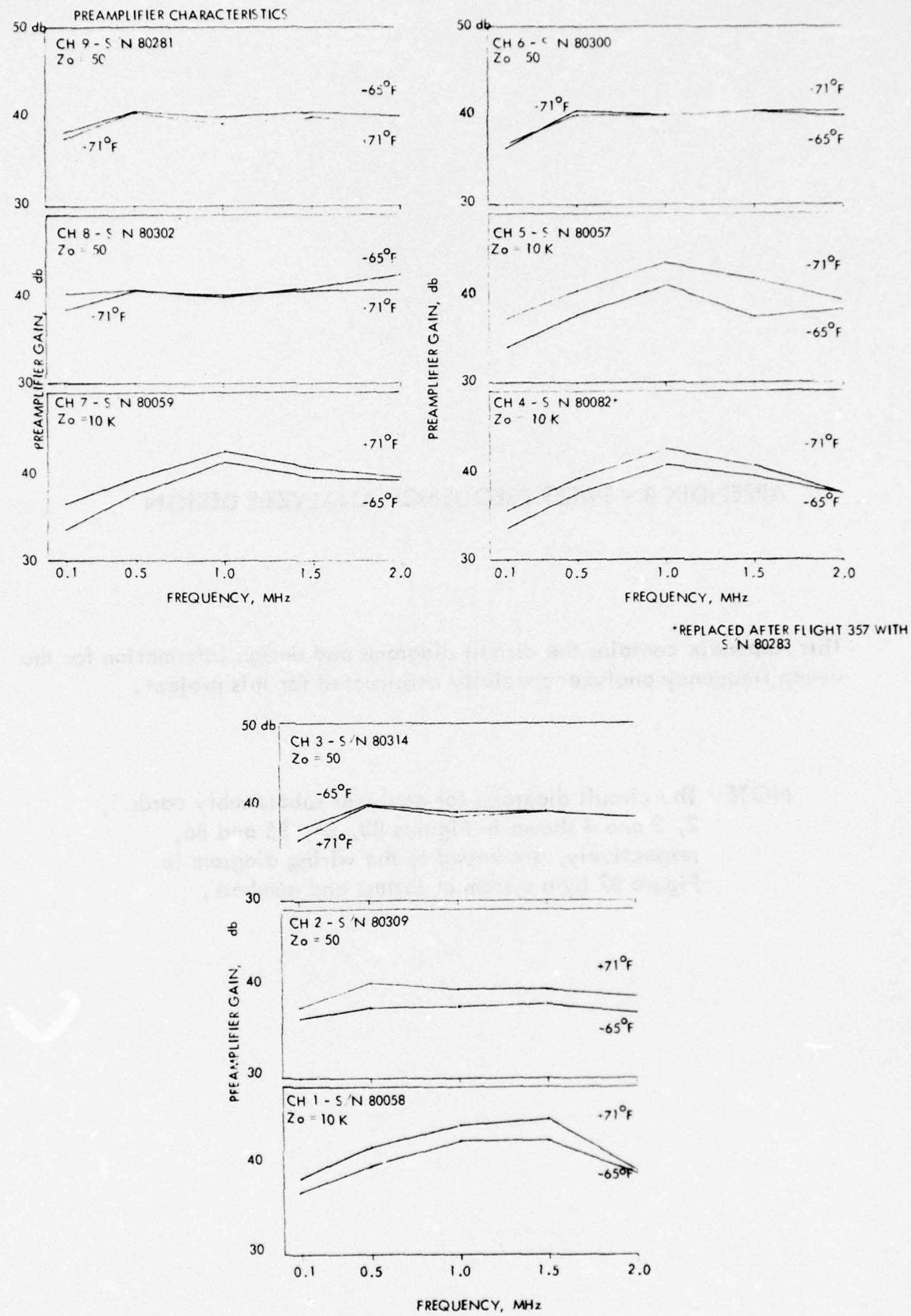


Figure A3. AE/SCG System Preamplifier Response at 71°F and -65°F.

APPENDIX B - SWEEP FREQUENCY ANALYZER DESIGN

This Appendix contains the circuit diagrams and design information for the sweep frequency analyzer specially constructed for this project.

NOTE: The circuit diagrams for analyzer subassembly cards 1, 2, 3 and 4 shown in Figures B3, B4, B5 and B6, respectively, are keyed to the wiring diagram in Figure B2 by a system of letters and numbers.

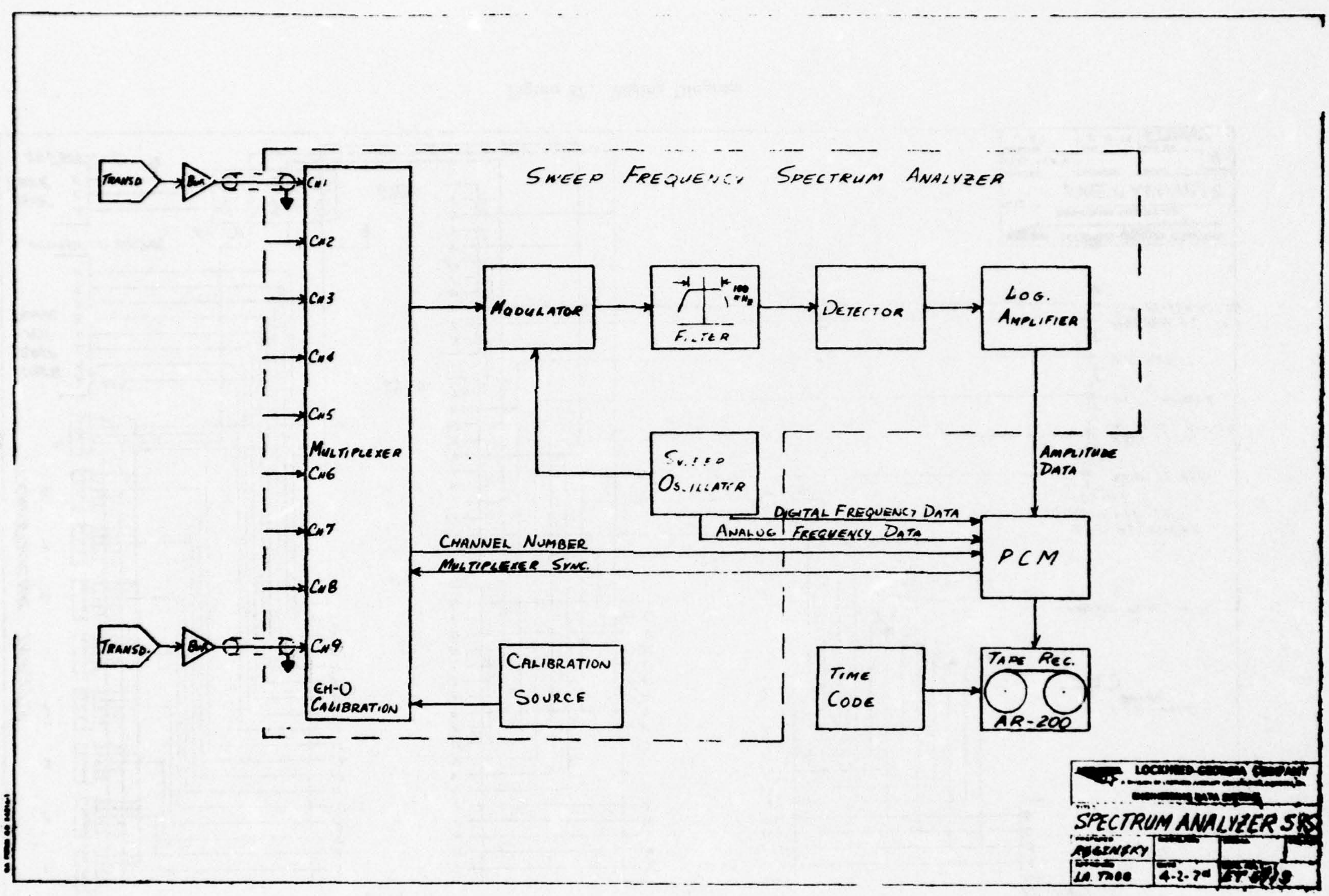


Figure B1. Block Diagram

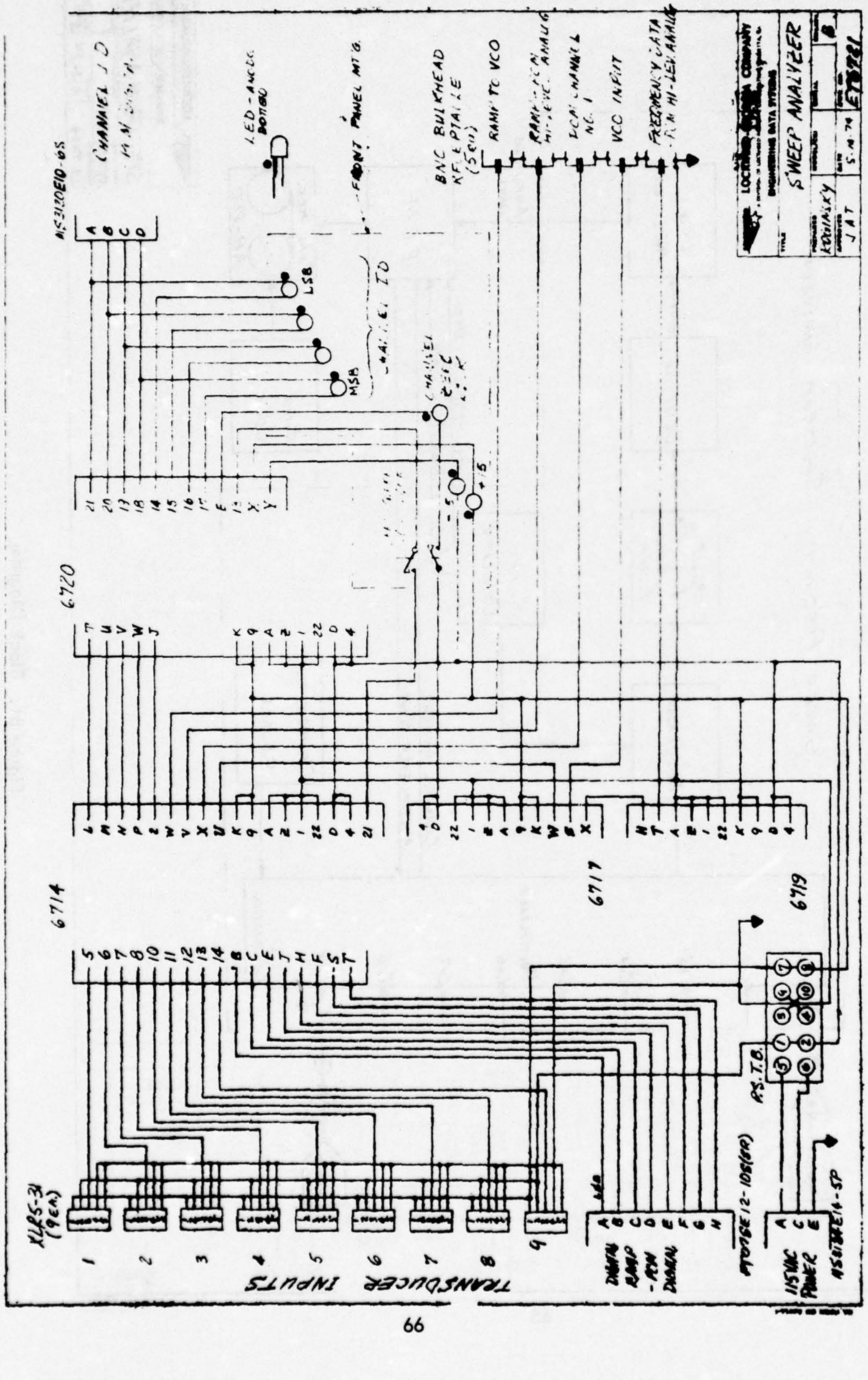


Figure B2. Wiring Diagram

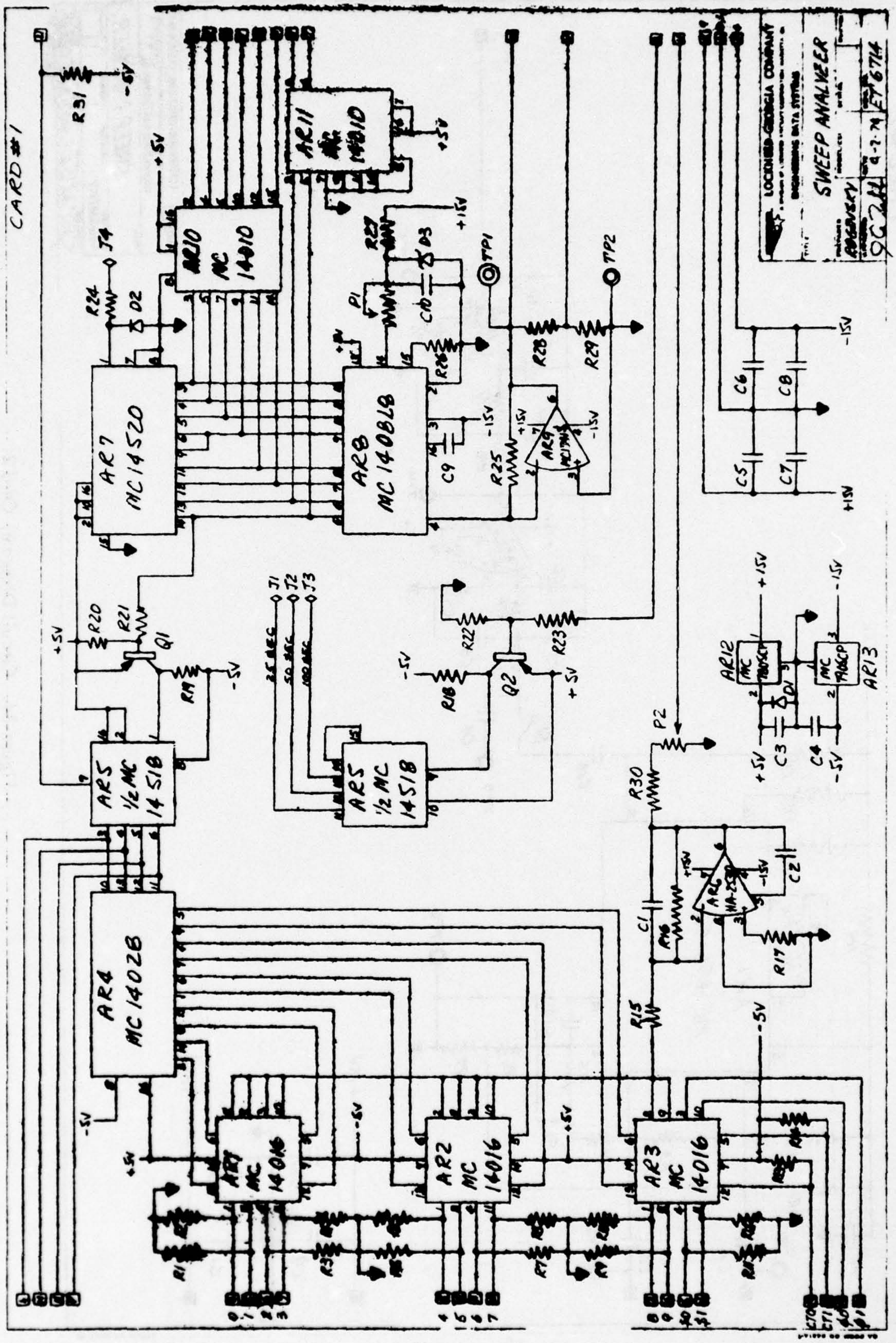


Figure B3. Circuit Diagram, Card 1

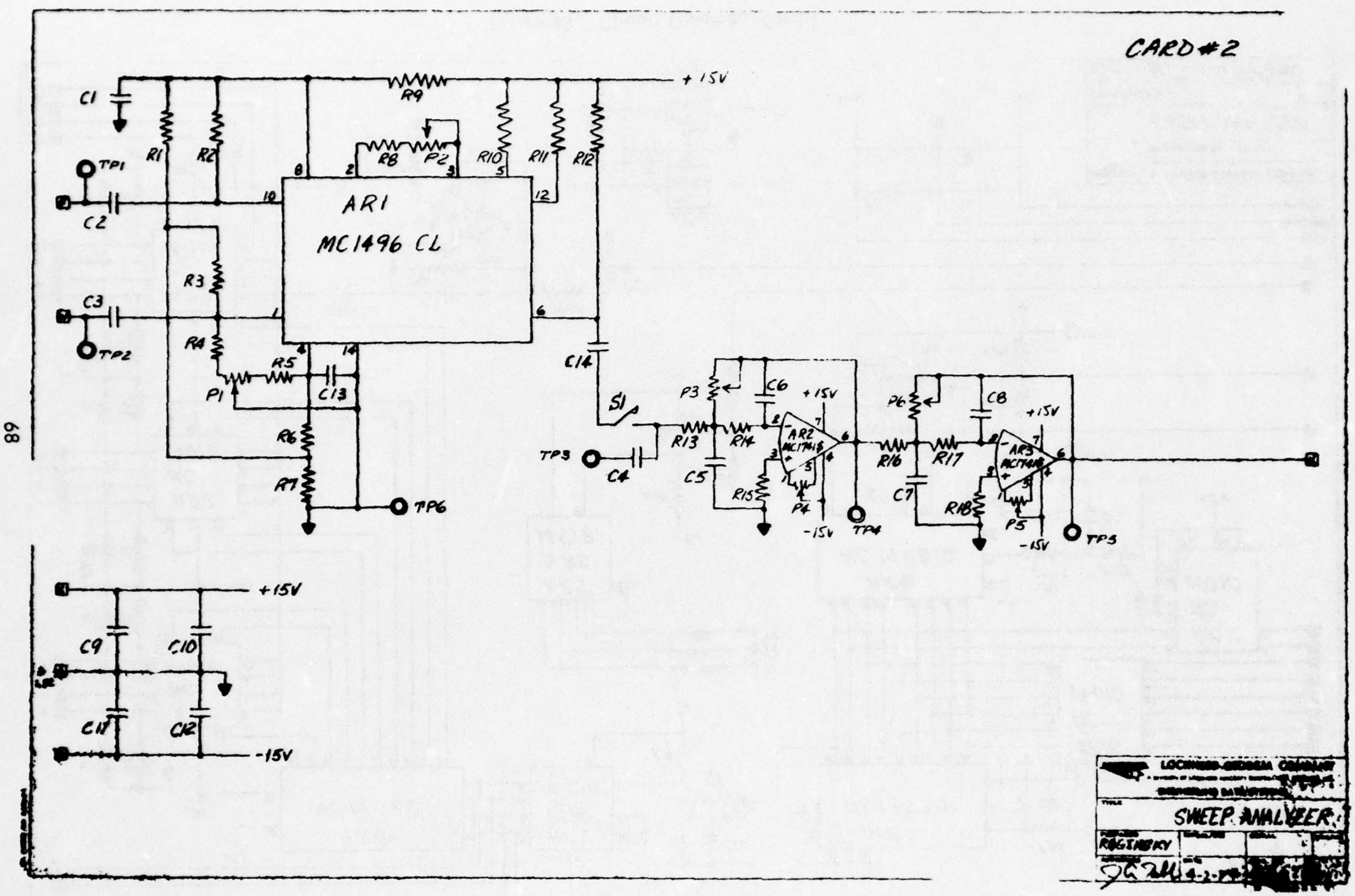


Figure B4. Circuit Diagram, Card 2

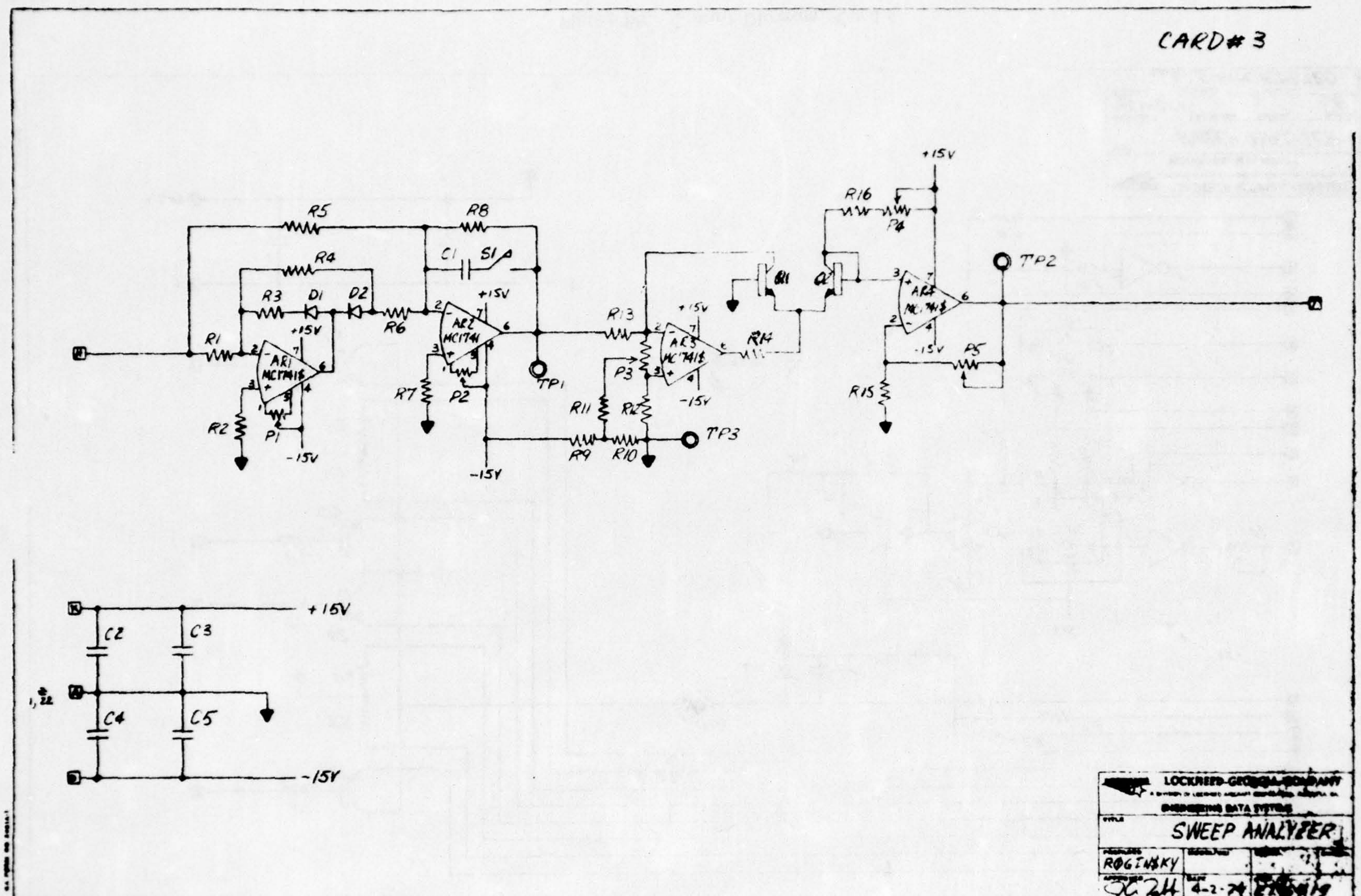


Figure B5. Circuit Diagram, Card 3.

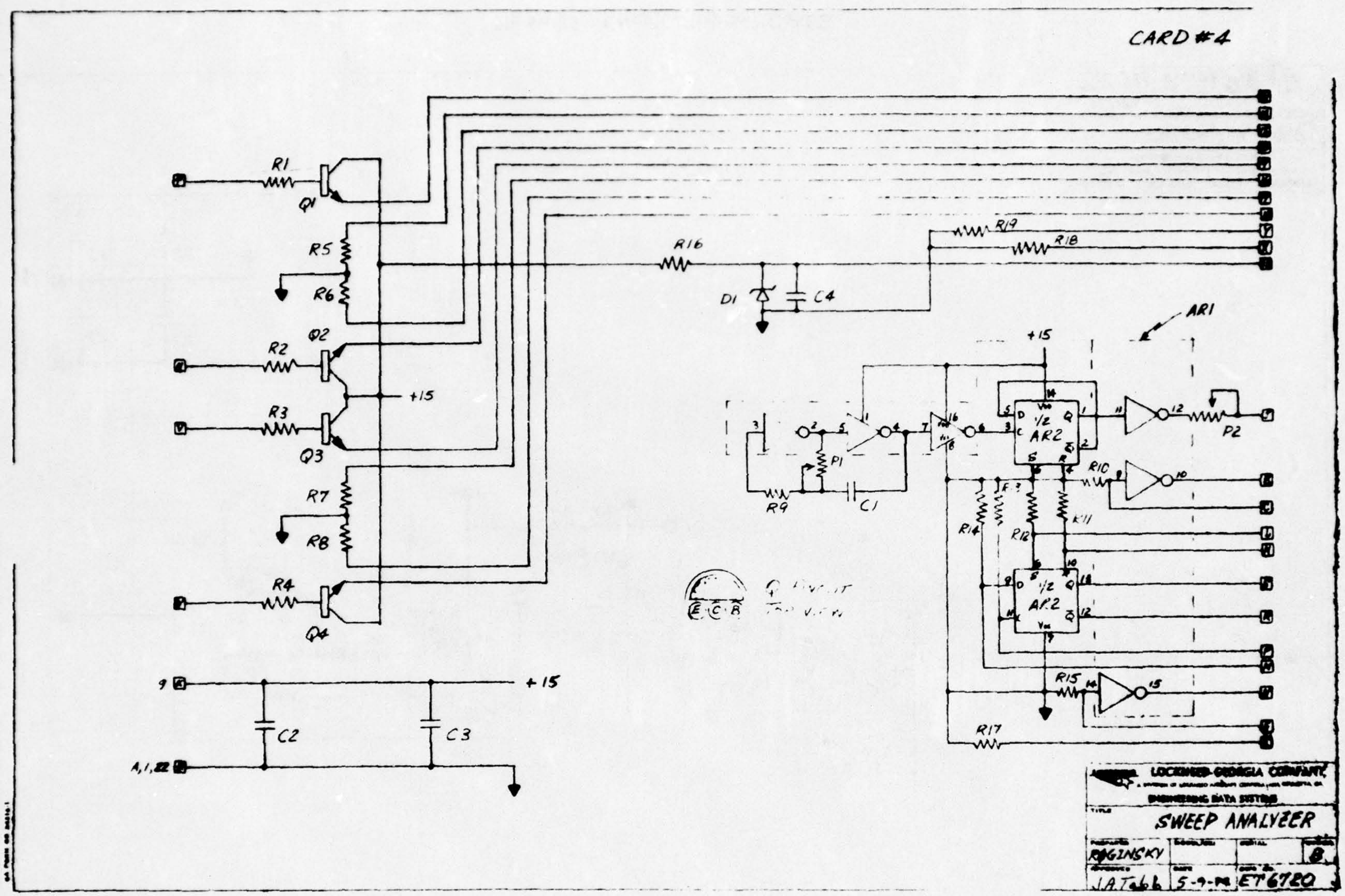


Figure B6. Circuit Diagram, Card 4

CARD #1

P/N	NOMENCLATURE	VALUE	RATING	SIZE
R1, R3 thru R12	CARBON RESISTOR	100Ω/5%	1/4 W	STD 1/4 W
R13, R.8 thru R18, R23, R31		39K/5%		
R15 & R16		2.2K/5%		
R30		470Ω/5%		
R20		27K/5%		
R21 & R24		10K/5%		
R22		7.6K/5%		
R25		4.99K/1%		
R26		2.61K/1%		
R17, R27 & R29		1K/5%		
R28		4.7K/5%		
P1	CARBON FILM POT	10K/5%	MOD 3897	3/8 IN SQ.
P2		100Ω/5%		
C1 & C2	EPOXY MOLDED CAP	TO BE SEL'TD	100V	CK05
C3 thru C8 & C10		.1μf/5%	100V	CK06
C9		22pf/5%	100V	CK05
D1	SILICON DIODE	1N5059	5A	.18x.15 IN
D2	SIGNAL DIODE	1N4446	500 MW	.195x.18 IN
D3	ZENER DIODE	1N4734A	1W	CASE 59 DO-41
Q1 & Q2	SILICON PNP TRS	2N5366	360 MW	EPOXY CKT.
AR6	HARRIS 2530 OPAMP	HA2530	—	TO99
AR9	MOTOROLA 1741S OPAMP	MC1741SCP	—	8 DIP

NOTE: R2 & C2 DELETED.

LOCKHEED-GEORGIA COMPANY
ENGINEERING DATA SYSTEMS
TITLE: FS/VS SWEEP ANALYZER
PREPARED: ROGINSKY
DATE: 4-[illegible]
DWG NO: ET6714A

CARD #1

P/N	NOMENCLATURE	VALUE	RATING	SIZE
AR1 thru AR3	MPX. SWITCH	MC14016	—	14 DIP
AR4	BCD TO DEC. CONV.	MC14028	—	16 DIP
AR5	BCD UP-CTR.	MC14518	—	16 DIP
AR7	BINARY UP-CTR.	MC14520	—	16 DIP
AR8	D/A CONV.	MC1408L8	—	16 DIP
AR10 & AR11	BUFFER	MC14010	—	16 DIP
AR12	+ 5V REG.	MC7805	1W-[illegible]	CASE [illegible]
AR13	- 5V REG.	MC7905	1W-[illegible]	CASE [illegible]

NOTES: 1. AR6 & AR9 TO BE MOUNTED ON INDIVIDUAL MOLEX SOCKETS.
2. JUMPERS J1 THROUGH J4 TO BE IN THE FORM OF PADS AND TO BE LOCATED NEAR EACH OTHER.

LOCKHEED-GEORGIA COMPANY
ENGINEERING DATA SYSTEMS
TITLE: SWEEP ANALYZER
PREPARED: ROGINSKY
DATE: 4-[illegible]
DWG NO: ET6714B

Figure B7 (1 of 3). Electrical Components Identification

CKT #2

P/N	NOMENCLATURE	VALUE	RATING	SIZE
R1	CARBON RESISTOR	82Ω, 5%	1/4W	STD 1/4W
R2		47Ω, 5%		
R3 & R6		120Ω, 5%		
R4, R5 & R10		10K/5%		
R7		1K/5%		
R8		100Ω/5%		
R9		1.5K/5%		
R11 & R12		3K/5%		
R13 & R14		1.91K/1%		
R15		470Ω/5%		
R16 & R17		75Ω/1%		
R18		68Ω/5%		
P1	CARBON FILM POT	100K/5%	MC 3389T	[illegible]
P2		1K/5%		
P3 THRU P6		10K/5%		
C1 THRU C4	EPOXY MOLD CAP	.1uf	100V	CK05
C5		2200pf		CK05
C6		560pf		
C7		3300pf		
C8		1500pf		
C9 THRU C14		.01uf		CK06
AR1	MOTOROLA MODULATOR	MC1496CL	—	14 DIP
AR2 & AR3	MOTOROLA OPAMP	MC1741SC	—	8 DIP
S1	SEE NOTE 2	—	—	

NOTES: 1. LAYOUT OF CAPACITORS C1, C3, C4 [illegible] TO PROVIDE MOUNTING OF EITHER [illegible] CAPACITORS OR SPRAGUE 6950A SIZE.

2. AR1 TO BE MOUNTED ON 16 ZIP SOCKET - UNUSED SET OF PINS TO SERVE AS S1 BY REMOVABLE JUMPER.

3. AR2 & AR3 SHALL [illegible]

LOCKHEED-GEORGIA COMPANY

ENGINEERING DATA SYSTEMS

TITLE: SWEEP ANALYZER

PREPARED: [illegible]

APPROVED: [illegible]

DWG NO: ET6[illegible]A

GA FORM GD 2481-1

Figure B7 (2 of 3). Electrical Components Identification

CARD #3

P/N	NOMENCLATURE	VALUE	RATING	SIZE
R1	CARBON RESISTOR	4.7K/5%	1/4 W	STD 1/4 W
R2	↓	3.3K/5%	↓	↓
R3,4,12,13,14,16		10K/5%		
R5 & R6		48.7K/1%		
R7		23.7K/1%		
R8		100K/5%		
R9 & R10		3.9K/5%		
R11		3.3M/5%		
R15	↓	470Ω/5%	↓	↓
P1 & P2	CARBON FILM POT	10K/5%	[illegible] 3389T	3/8 [illegible]
P3 & P4	↓	1M/5%	↓	↓
C1 THRU C5	EPOXY MOLDED CAP	.1uf/5%	[illegible]	[illegible]
D1 & D2	SIGNAL DIODE	1N4448	500mW	[illegible]
Q1 & Q2	EPOXY MOLDED TRAN	2N2417	360mW	[illegible]
AR1,3 & 4	MOTOROLA OPAMP	MC1741SC		8 DIP
AR2	↓	MC1741C	—	↓
S1	SEE NOTE 2	—		

NOTES: 1. AR[illegible] TO BE MOUNTED [illegible]
2. AR2 TO BE MOUNTED ON 14 DIP [illegible] [illegible] [illegible]
3. AR[illegible] & AR4 TO [illegible]

LOCKHEED-GEORGIA COMPANY
ENGINEERING DATA SYSTEMS
[illegible] SWEEP ANALYZER

CARD #4

P/N	NOMENCLATURE	VALUE	RATING	SIZE
R1 THRU R4	CARBON RESISTOR	27KΩ	1/4 W	STD. 1/4 W
R5 THRU R8	CARBON RESISTOR	220Ω	1/4 W	STD. 1/4 W
R9	CARBON RESISTOR	47KΩ	1/4 W	STD. 1/4 W
P1 & P2	"CERMET" CARBON	50KΩ	[illegible] 3389T	3/8 [illegible]
C1	"MONOBLOCK" CERAMIC	22pf	100V, 10%	CKC5K
C2 & C3	"MONOBLOCK" CERAMIC	.1uf	100V, 20%	CK[illegible]M
R10 THRU R15	CARBON RESISTOR	39KΩ	1/4 W	STD. 1/4 W
Q1 THRU Q4	EPOXY TRANSISTOR	2N2417	360 mW	[illegible]
AR1	INVERTER BUFFER	MC14049 [illegible]	—	16 DIP
AR2	DATA FLIP-FLOP	MC14013CP	—	14 DIP
R16	CARBON RESISTOR	2.2KΩ	1/4 W	STD. 1/4 W
D1	SILICON ZENER	1N4732A	1W, 5%	[illegible]
C4	"MONOBLOCK" CERAMIC	.1uf	100V, 20%	CK06M
R17 THRU R19	CARBON RESISTOR	680Ω	1/4 W	STD. 1/4 W

LOCKHEED-GEORGIA COMPANY
ENGINEERING DATA SYSTEMS
SWEEP ANALYZER
ROGINSKY
JA 7266 | 5-5-70 | ET6720A | A

Figure B7 (3 of 3). Electrical Components Identification

APPENDIX C - FLIGHT/CALIBRATION DATA REDUCTION

This Appendix contains data reduced from the pre-flight calibration and AE/SCG flight test recordings, presented in graphs or tabular form, used in the analysis of the data.

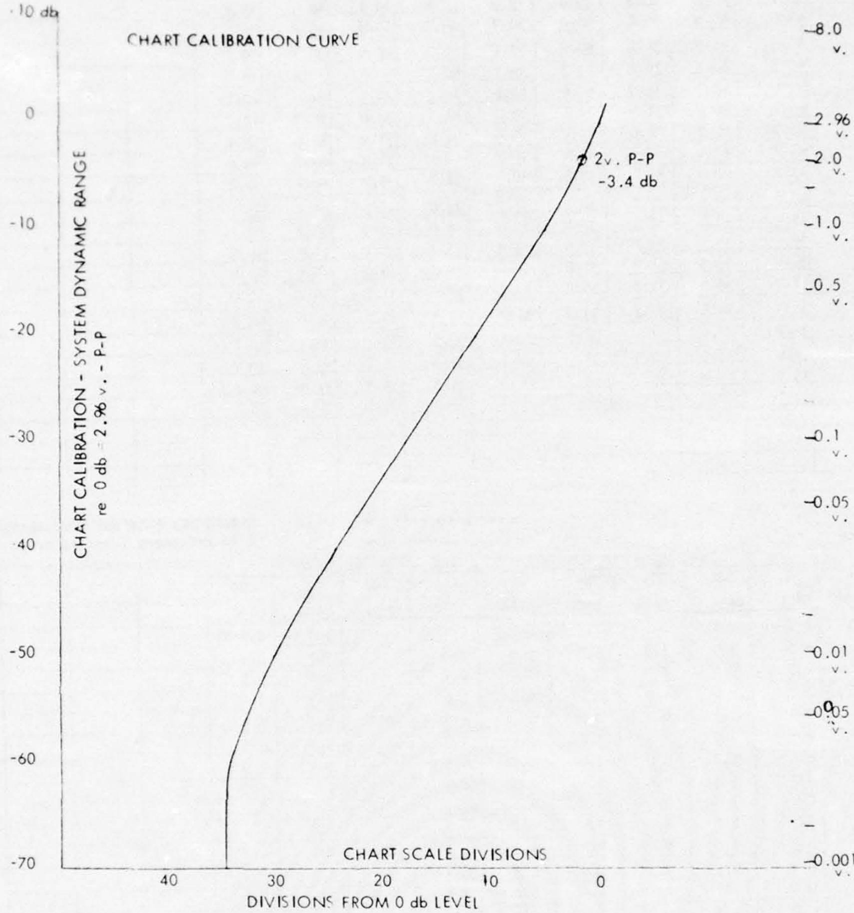


Figure C1. Calibration Curve for the Strip-Chart Recording Readouts of the Flight Magnetic Tape Recorder.

CALIBRATION
SCG SYSTEM

LOCKHEED AIRCRAFT CORPORATION
Georgia Division

SYSTEM GAIN (AMPLIFIER GAIN - CABLE ATTENUATION)
REF. WHITE NOISE INPUT TO PREAMPLIFIER

FREQUENCY KHz	NET SYSTEM GAIN, db								
	CH1	CH2	CH3	CH4	CH5	CH6	CH7	CH8	CH9
100 KHz	40.0 db	40.0 db	38.3 db	40.0 db	40.0 db	38.5 db	40.7 db	40.7 db	41.5 db
138	39.8	40.0	37.3	40.0	39.4	38.9	39.6	40.5	40.6
176	39.6	39.7	36.3	39.8	38.8	39.0	38.6	40.5	39.8
214	39.6	39.7	35.8	39.6	38.6	39.1	38.3	40.4	39.4
252	39.8	40.0	36.0	39.8	38.8	39.4	39.1	40.3	39.5
290	40.0	40.0	36.0	40.0	38.9	39.7	39.8	40.2	39.5
328	40.5	40.3	36.2	40.2	39.1	39.9	40.6	39.3	39.7
366	40.8	40.5	36.6	40.3	39.3	40.2	40.5	38.2	39.9
404	41.1	40.6	36.9	40.6	39.6	40.7	39.3	37.5	40.1
442	41.7	40.8	37.4	41.1	40.0	41.5	38.8	38.7	40.3
480	41.9	41.7	38.0	41.4	40.5	41.2	37.9	40.0	41.1
518	41.7	41.1	38.4	41.0	40.2	40.1	38.2	40.7	41.6
556	41.1	40.4	38.4	39.8	38.8	38.8	39.2	40.7	41.8
594	40.2	39.5	37.7	38.7	37.7	37.8	39.2	40.6	42.1
632	39.4	38.6	36.7	37.8	36.6	36.6	38.8	40.2	42.7
670	38.7	37.7	35.8	37.0	35.5	35.7	38.4	39.8	42.8
708	38.0	36.5	34.8	36.0	34.0	34.7	38.0	39.5	42.3
746	37.2	35.9	33.8	35.2	33.2	33.7	37.7	39.2	41.6
784	36.3	34.8	33.2	33.9	31.7	32.4	37.2	38.8	40.8
822	35.4	33.9	32.2	32.8	30.2	31.9	36.7	38.3	40.2
860	34.6	33.0	30.8	31.8	28.7	31.8	36.3	37.8	39.7
898	33.8	31.9	29.3	30.8	27.0	34.3	35.8	37.3	39.2
1050	30.3	28.3	25.0	26.2	26.2	31.6	34.3	36.0	35.0

CALIBRATION
TRANSDUCERS

LOCKHEED AIRCRAFT CORPORATION
Georgia Division

TRANSDUCER RESPONSE NONLINEARITY.
0 db reference level at Response Peak

FREQUENCY KHz	CORRECTION FOR NONLINEARITY OF TRANSDUCER RESPONSE								
	CH1	CH2	CH3	CH4	CH5	CH6	CH7	CH8	CH9
TRANSDUCER	8A06	8A34	8A24	8A03	8A46	8A28	8A36	8A04	8A40
100 KHz	15.7 db	6.7 db	7.3 db	7.2 db	9.8 db	6.8 db		9.8 db	9.5 db
138	8.0	1.5	1.3	3.2	5.2	1.6		3.7	1.6
176	4.3	0.4	2.0	0.6	5.1	0.1		1.0	1.1
214	2.5	1.0	4.3	0.2	6.1	0.5		1.3	3.5
252	0.2	2.1	8.0	0.8	6.0	2.5	20.5	4.7	7.5
290	2.3	3.0	7.0	3.0	3.1	4.8	13.9	9.2	8.5
328	6.3	3.1	6.8	3.8	1.7	6.0	9.1	9.9	7.9
366	8.2	3.0	7.6	4.6	0.5	6.1	5.5	9.7	7.3
404	8.9	3.2	9.3	5.2	0.1	6.7	2.8	8.7	7.5
442	9.3	1.0	8.9	2.4	0.6	6.8	0.8	6.8	7.8
480	8.7	1.5	8.2	2.0	4.1	5.9	0.1	6.5	9.7
518	8.3	2.5	7.5	1.6	4.8	5.5	1.5	7.0	10.7
556	9.0	3.5	6.8	1.4	5.2	6.1	3.2	7.5	11.0
594	9.4	3.5	6.8	1.8	5.2	6.7	3.5	6.0	11.3
632	9.0	5.2	8.2	2.4	4.5	6.8	5.5	6.7	11.5
670	8.8	6.5	7.7	3.2	5.2	6.8	6.7	7.8	11.5
708	9.0	7.2	8.3	3.6	6.8	7.1	7.0	9.0	11.3
746	9.5	7.2	8.6	3.6	6.8	9.0	8.5	9.0	10.9
784	10.6	8.5	9.1	4.8	7.2	9.9	8.9	9.7	11.4
822	11.4	9.2	9.9	5.6	9.2	10.8	9.3	10.1	11.2
860	12.2	9.8	10.9	6.2	11.5	12.0	10.5	11.2	11.3
898	12.7	10.4	12.1	6.6	12.8	12.0	14.5	14.5	12.7
1050	16.5	11.7	16.5	10.6	15.5	15.8	20.0	17.8	17.1
0 db ref. level at	250 KHz	150 KHz	150 KHz	200 KHz	400 KHz	200 KHz	475 KHz	200 KHz	150 KHz
re 1v/microbar →	-79.6 db	-79.5 db	-78.7 db	-73 db	-79.2 db	-77.2 db	-68.5 db	-77.3 db	-76.5 db

Form 1876A-1

TABLE C1 (UPPER). SYSTEM GAIN FOR EACH CHANNEL.
TABLE C2 (LOWER). RESPONSE NONLINEARITY FOR EACH TRANSDUCER IN TERMS OF DECIBELS BELOW RESPONSE PEAK.

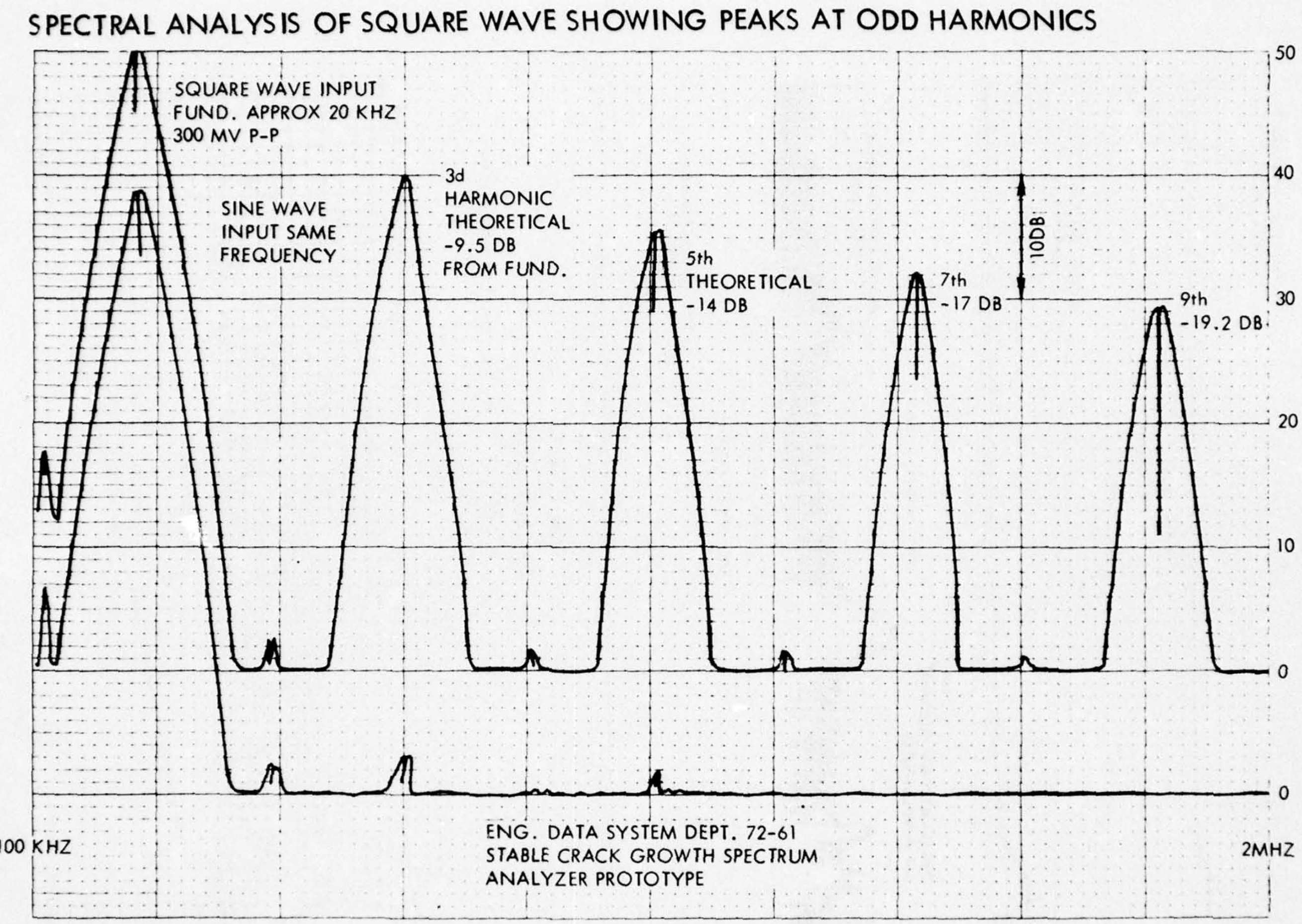


Figure C2. Spectral Analysis of Square Wave vs Sine Wave Input to the Sweep Frequency Analyzer.

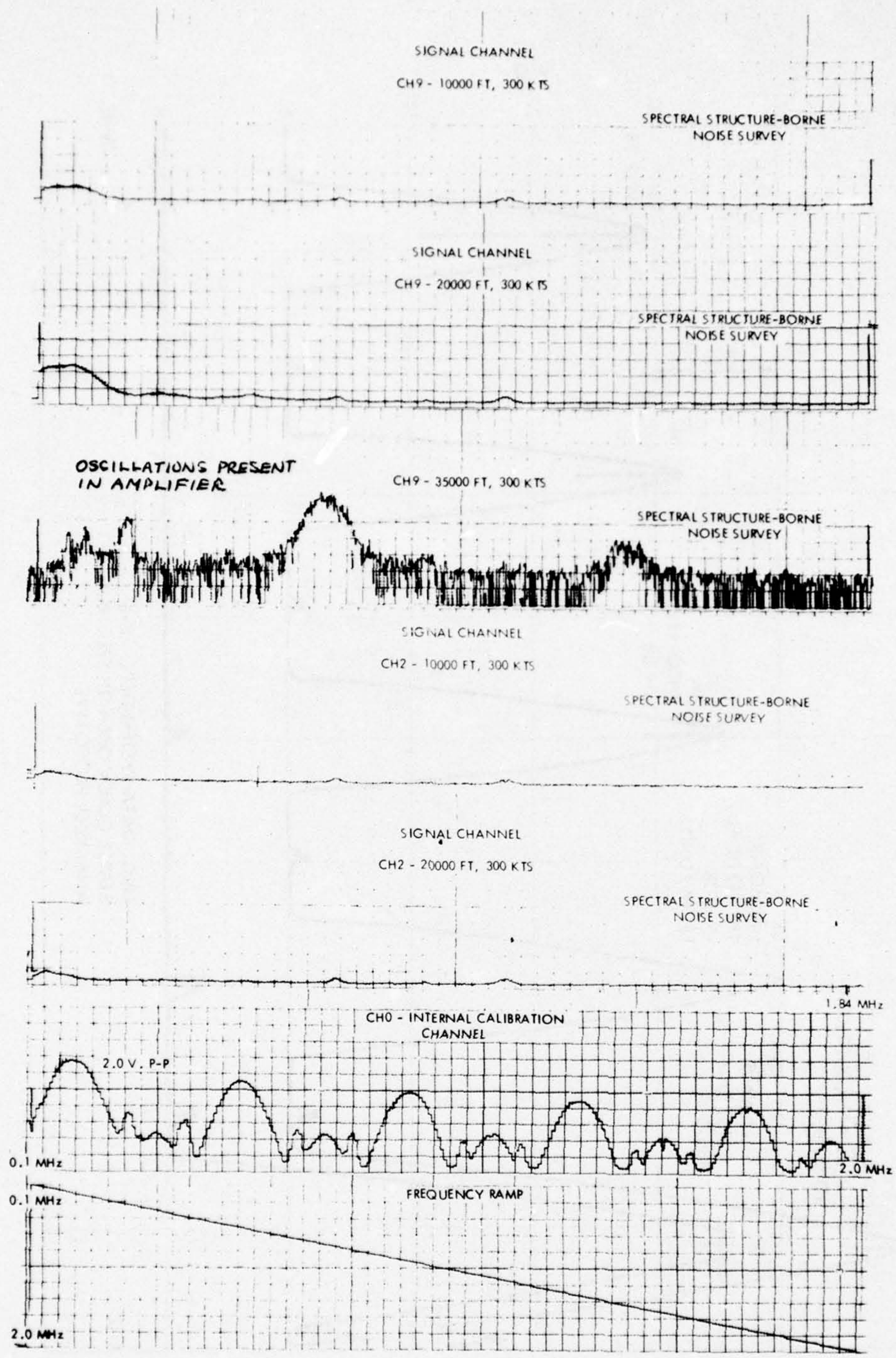


Figure C3 (1 of 2). Spectral Structure-Borne Flight Noise Received at Several Channels at 10,000, 20,000, and 35,000 Feet Altitudes.

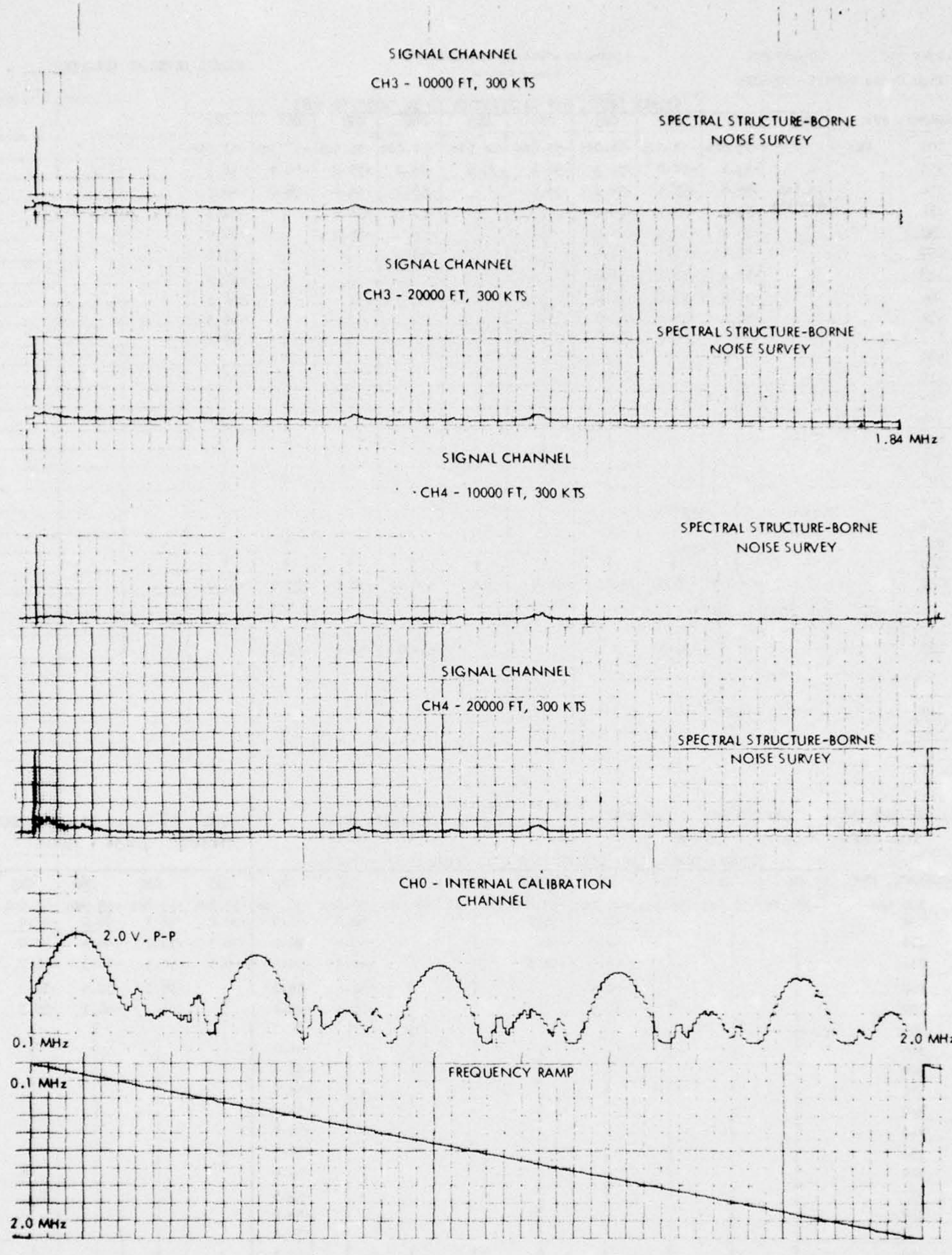


Figure C3 (2 of 2). Spectral Structure-Borne Flight Noise Received at Several Channels at 10,000, 20,000, and 35,000 Feet Altitudes.

FLIGHT 356 10,000 FT. LOCKHEED AIRCRAFT CORPORATION Georgia Division SIGNAL LEVEL AT ANALYZER

Time Codes 050835 - 051815

FREQUENCY, KHz	SIGNAL LEVEL, REF CALIBRATION 0 db LEVEL (2.96V) CH0	CH1	CH2	CH3	CH4	CH5	CH6	CH7	CH8	CH9
100 KHz		-55.8db	-54.0db	-54.0db	-55.8db	-56.6db	-56.2db	-55.5db	-55.5db	-51.3db
138		-53.4	-51.8	-56.3	-56.6	-58.0	-55.0	-55.8	-56.4	-48.5
176	-3.4db	-55.8	-53.4	-56.6	-58.0		-56.3	-56.4	-58.0	-48.5
214	@195KHz	-55.0	-55.8	-56.6			-56.3	-56.4		-51.3
252		-56.3	-56.3	-56.6			-56.6	-58.0		-55.0
290		-56.4	-56.3	-56.6			-58.0			-55.8
328		-56.4	-56.3	-56.6						-56.4
366		-56.4	-55.8	-55.8						-56.3
404		-58.0	-56.3	-56.3						-56.4
442			-58.0	-58.0						-58.0
480										
518										
556										
594										
632										
670										
708										
746										
784										
822										
860										
1050		-58.0	-58.0	-58.0	-58.0	-58.0	-58.0	-58.0	-58.0	-58.0
120			-51.3				-54.0	-54.6	-52.5	
348		-55.8								

FLIGHT 357 20,000 FT 340 KTS LOCKHEED AIRCRAFT CORPORATION Georgia Division SIGNAL LEVEL AT ANALYZER

TIME CODES 095352 - 102940

FLIGHT 357† 10,000 FT (DESCENDING)

TIMECODES 104554 - 110622

FREQUENCY, KHz	SIGNAL LEVEL, REF. CALIBRATION 0 db LEVEL (2.96 v P-P) CH1	CH2	CH3	CH4	CH5	CH6	CH7	CH8	CH9	FLIGHT 357† CH5	CH7	CH8	CH9*
100 KHz	-58.5db	-58.5db	-58.5db	-47.2db	-53.8db	-58.5db	-58.5db	-53.9db	-38.7db	-53.8db	-53.8db	-53.9db	-51.0db
138				-42.8	-50.2			-48.0	-36.2	-50.2	-52.4	-48.0	-47.0
176				-49.8	-56.3			-48.4	-36.2	-56.3	-56.0	-48.4	-47.0
214				-49.5	-58.5			-46.3	-36.4	-58.5	-56.3	-46.3	-48.7
252				-52.4				-51.4	-44.0		-58.5	-51.4	-55.0
290				-58.5				-58.5	-49.0			-58.5	-58.5
328									-52.5				
366									-51.0				
404									-51.0				
442									-52.5				
480									-54.2				
518									-53.8				
556									-52.5				
594									-51.0				
632									-52.5				
670									-54.2				
708									-55.2				
740									-58.5				
1050	-58.5	-58.5	-58.5	-58.5	-58.5	-58.5	-58.5	-58.5	-58.5	-58.5	-58.5	-58.5	-58.5

*CH 9 indications for this run were made during landing gear cycles at optimized altitude and speed (<10,000 feet).

†This listing contains only those channels which registered noise signals.

TABLE C3 (UPPER). STRUCTURE-BORNE FLIGHT NOISE LEVELS AT THE ANALYZER FOR FLIGHT #356 AT 10,000 FT, O DECIBELS EQUIVALENT TO 2.96 VOLTS.

TABLE C4 (LOWER). STRUCTURE-BORNE FLIGHT NOISE LEVELS AT THE ANALYZER FOR FLIGHT #357, O DECIBELS EQUIVALENT TO 2.96 VOLTS.

FLIGHT 359 10,000 FT.

Flight Codes 064515 - 065755

LOCKHEED AIRCRAFT CORPORATION
Georgia Division

SIGNAL LEVEL AT ANALYZER

	SIGNAL LEVEL, REF CALIBRATION 0 db LEVEL (2.96V)									
FREQUENCY, KHz	CH0	CH1	CH2	CH3	CH4	CH5	CH6	CH7	CH8	CH9
100 KHz		-59.0db	-57.3db	-59.0db	-56.7db	-56.6db	-56.4db	-56.4db	-59.0db	-45.0db
138			-56.0		-55.4	-56.6	-56.0	-56.4		-41.5
176	-3.4db		-56.6		-57.8	-59.0	-57.8	-57.8		-42.4
214	@195KHz		-57.8		-58.2		-59.0	-59.0		-44.7
252			-59.0		-59.0					-51.3
290										-55.0
328										-56.0
366										-55.8
404										-56.0
442										-57.0
480										-58.2
518										-56.4
556										-56.0
594										-56.0
632										-56.6
670										-57.8
708										-58.2
746										-59.0
784										
822										
860										
1050		-59.0	-59.0	-59.0	-59.0	-59.0	-59.0	-59.0	-59.0	-59.0
120					-53.6		-55.0			
576										-55.4

FLIGHT 359 20,000 FT

TIME CODES 071000 - 071920

LOCKHEED AIRCRAFT CORPORATION
Georgia Division

SIGNAL LEVEL AT ANALYZER

	SIGNAL LEVEL, REF CALIBRATION 0 db LEVEL (2.96V)									
FREQUENCY, KHz	CH0	CH1	CH2	CH3	CH4	CH5	CH6	CH7	CH8	CH9
100 KHz		-55.8db	-54.4db	-53.0db	-54.8db	-57.5db	-58.2db	-56.0db	-58.5db	-41.3db
138		-55.8	-51.3	-53.7	-49.5	-56.4	-56.4	-55.7		-39.0
176	-3.4db	-56.4	-53.5	-55.5	-54.6	-58.0	-58.0	-56.8		-38.0
214	● 195	-56.0	-56.0	-56.0	-57.5	-58.0	-58.0	-57.6		-41.4
252		-56.6	-56.6	-56.4	-58.2	-58.5	-58.5	-57.3		-48.0
290		-56.3	-56.3	-56.0	-58.5			-57.5		-52.0
328		-57.5	-57.5	-56.4				-56.8		-53.5
366		-58.5	-58.5	-57.3				-57.3		-53.0
404				-58.2				-58.2		-53.0
442				-58.5				-58.5		-54.2
480										-56.0
518										-55.0
556										-54.0
594										-53.5
632										-55.8
670										-56.0
708										-56.0
746										-56.0
784										-58.2
822										-58.5
860										
898										
936										
1050		-58.5	-58.5	-58.5	-58.5	-58.5	-58.5	-58.5	-58.5	-58.5
120			-50.5	-52.5			-56.6	-55.8		
158					-53.5					

TABLE C5 (UPPER). STRUCTURE-BORNE FLIGHT NOISE LEVELS AT THE ANALYZER FOR FLIGHT #359 AT 10,000 FT, O DECIBELS EQUIVALENT TO 2.96 VOLTS.

TABLE C6 (LOWER). STRUCTURE-BORNE FLIGHT NOISE LEVELS AT THE ANALYZER FOR FLIGHT #359 AT 20,000 FT, O DECIBELS EQUIVALENT TO 2.96 VOLTS.

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